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JOURNAL OF THE AERONAUTICAL SCIENCES

VOLUME 10 JANUARY, 1943 NUMBER 1

Flight Testing for Performance and Stability

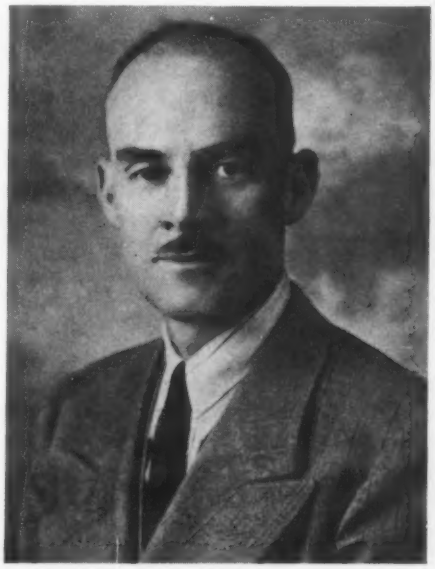
*The Sixth Wright Brothers Lecture**

EDMUND T. ALLEN†
Boeing Aircraft Company

Introduction

THIRTY-NINE YEARS AGO today, Orville Wright and his brother Wilbur made successful flight tests which demonstrated that they had solved the two most fundamental and essential of flight problems, namely, performance and stability. This momentous event, which was little noticed at the time, changed the destiny of man. It has taken 39 years for the full flood of its importance to touch every country on earth. At that time, however, back in 1903, this test flight may have appeared to many as a fantastic stunt. It was by no means this. It was rather the initial stage of a scientific technique that has become a corner stone in airplane design. Only a few friends of the Wright Brothers knew of the painstaking years of experimentation which preceded that first flight or of the painstaking test flying that followed it in order to measure the characteristics of each new improvement and to analyze what was still needed to provide stability, control and performance. Fewer still were aware of the problems of thrust horsepower available and aerodynamic power required which had to be solved successfully before flight was possible. As the Wright Brothers added their developments, such as the front elevator, wing warping, and the vertical rear fins, they were utilizing the most modern aerodynamic technique in transferring from their wind tunnel to flight test each increment of aerodynamic knowledge as it became available from the laboratory to test in practical use. They spent all the years from 1900 to 1908 testing, testing, testing, before they sold any airplanes. Here was flight testing on as high a scale of scientific intent as it has ever reached since then.

During the 39 years since this first flight test, each new airplane design in the long series of steps upward toward air power has been an exploration into untried territory and a measurement of design progress. That progress has been principally along three parallel lanes: power to drive the propellers, structure to increase strength and reduce weight, and aerodynamics to decrease drag and increase stability and control. Without the truly amazing progress of any one of these three developments, the modern airplane would be utterly impossible. The modern aircraft engine with its thousands of horsepower weighs per horsepower but a small fraction of the weight of the engines of 30 years ago. The modern high aspect ratio monoplane wing so essential to aerodynamic refinement could not have been built without the intensive struc-



Edmund T. Allen.

* Presented before the Institute of the Aeronautical Sciences at Columbia University, New York, December 17, 1942.

† Director of Flight and Aerodynamics.

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tural research and development of the past 20 years. And finally, aerodynamics has found out how to decrease air resistance to a phenomenally low level and has, in addition, tied together the three lanes of development into a mathematical pattern that determines the performance obtainable with any combination of power and structure.

Aerodynamics has developed by means of two types of testing: wind tunnel and flight. The wind tunnel has been an extremely important tool in design progress since the opening years of this century. The Wrights investigated aspect ratio and camber of wings and determined the L/D of a lifting surface in a small wind tunnel of their own. In the early 1920's, wind-tunnel technique was rapidly developed with the exploration of thousands of new airfoils. Complete models of new airplanes were tested to determine, before flight, their performance and stability. At this stage, inaccuracies in technique gave rise to unexplained discrepancies between the wind tunnel and flight test which led many designers to believe that the wind tunnel was not a satisfactory means for the prediction of airplane characteristics and that flight testing of the finished airplane was the only way of refining a design. In the early 1930's, wind-tunnel development was greatly accelerated by new analysis of flow phenomena and by greatly improved techniques that resulted in a revolutionary change in the concepts of designers. This wind-tunnel renaissance was followed by a new flight-testing technique that opened up a vista of accuracy never before believed possible.

Each of these methods, wind-tunnel test and flight test, have been used in two ways: first, as a cut-and-try exploration to find an optimum design; second, as a quantitative measure of performance and stability. The cut-and-try technique has recently been elaborated in scope and extent both in flight and in the wind tunnel. It has become an indispensable part of design as the more inspirational or artistic phases of it have given way to empirical phases and, finally, to the physically derived equations of fluid mechanics.

Thus, it has come about that, whereas ten years ago it was customary to build a simple wind-tunnel model to test in a single series of measurements to determine performance and stability and then to construct a highly experimental airplane on which the cut-and-try fine points of drag and control could be worked out in a long series of flights, it is now customary to reverse this process and do our cutting and trying in a most elaborate and lengthy series of wind-tunnel tests in which an optimum configuration is finally established. The flight test then tends to become a series of quantitative measurements of performance, stability and controllability plus a large number of minor measurements and final adjustments. This change that has taken place in a relatively short period has occurred at the same time as, and parallel to, a change in the whole approach to flight testing. This approach is what I would like to term an "aerodynamic" approach, since it deals largely with coefficients of the aerodynamic forces and moments acting on the airplane.

When the preflight stages of design and wind tunnel test have been thoroughly completed little cut-and-try exploration remains to be done in the flight-test stage. This is probably more completely true on large airplanes than on small ones. Several large airplanes have recently been developed in this manner so thoroughly that no major alterations were necessary to secure predicted performance, stability and controllability. The flight testing on such airplanes is no less elaborate with regard to instrumentation and accuracy than on former ones—rather the reverse. But its function now is to measure the coefficients involved in performance and stability so that this conformation of design data can be used more effectively in future design, as well as to give a detailed knowledge of the characteristics of the airplane at hand. This is perhaps an idealized picture. Actually, a good deal of cut-and-try experimentation must be done in flight to prove that the results found in a wind tunnel are satisfactory in full scale and under dynamic conditions. It has been found possible frequently to flight-test a part of an airplane on an older standard model prior to the completion of the airplane for which the part was designed. In many cases, this can be done with a great saving in time and safety. If this part is a control or stabilizing surface, for instance, it can be flight-tested either to full scale or near full scale on a standard model airplane. In this way, the cut-and-try experimentation required after wind-tunnel tests can be accomplished long before the experimental airplane is ready to fly.

The use of parallel testing in flight and wind tunnel has proved to be of great value. The wind-tunnel tests cannot always be expected to give definite quantitative answers, but they can be used to expedite the flight program in many ways. Important variables can usually be established in the tunnel and the approximate effects of small changes in each variable measured. Thus, a basis for rational analysis of the airplane during the flight-test program is available to prevent the making of wasteful or dangerous exploratory changes in the search for improvement. A satisfactory design answer to flight problems may thus be obtained in a shorter time than would be possible without the tunnel guidance. The wind-tunnel tests can, however, in nowise replace the flight test and it is actually only an extremely useful tool of the airplane designer.

The immediate objective of flight testing is usually to discover how close the designer has come to achieving a satisfactory design as well as the desired performance. If the testing personnel has the design point of view and a scientific background, it may look upon the airplane as a mathematical equation, performing in accordance with certain definite laws governing such

factors as power required and stability derivatives. A personnel of this kind can furnish the designing engineers with data in a particular form and of such accuracy that design progress is accelerated and the whole panorama of design regarded in a new light.

There are two primary divisions of flight testing, namely, performance and stability. Performance includes the vitally important measurements of speed, range, take-off, climbing ability and ceiling, while stability in its larger sense may be understood to include the flying qualities of an airplane, its freedom from dangerous diving or stalling tendencies, and the general smoothness and ease of its flight control.

Performance

The degree of accuracy attainable in measuring performance has been a highly controversial source of irritation to both commercial users of aircraft interested in economy and competitive performance and now, more especially, to the military services where such accuracy may day after day be a matter of life and death. An accurate flight test requires four accurate factors:

(1) Instrumentation so that all variables can be measured.

(2) Technique of flying accurately so that what is measured will be representative of the true optimum performance of the airplane.

(3) Steady air to fly in, with no vertical currents and a reasonable temperature pattern.

(4) A mathematically correct analytic method to apply to the flight-test data.

One difficulty of obtaining accuracy arises from the large number of variables involved in performance, many of which are disregarded or unknown at either the original testing station or at the far-distant air base where aircraft are sent out on their missions. As an example of what happens in ordinary flight testing, an aircraft company may design an airplane with great attention to detailed aerodynamics and wind-tunnel testing. From this analysis, it may appear that the airplane will have a drag such that its maximum speed at sea level is 300 m.p.h., its ceiling is 30,000 ft. and its maximum range is 3,000 miles. When the airplane is flown, a test pilot may fly it across the speed course at 320 m.p.h., climb to 35,000 ft., and get a minimum fuel consumption for long-range operation that figures out to give 15 per cent more range than the wind-tunnel tests indicated could be expected. When the user gets the airplane, his test pilot will find that he is unable to get over 280 m.p.h. high speed, a much lower ceiling and a poor range. If Army, Navy and British are users and each gets one of these airplanes for trials, the chances are that they will differ markedly in their findings because their testing techniques and reduction methods are different. If they test first in winter and later in summer, the results, even after reduction to so-called standard conditions, will be quite different. None of these tests will have measured the basic drag and propeller efficiencies from which a true picture of the potentialities of the design may be obtained and with which the use of the airplane in a military campaign can be accurately planned. As a result of this confusion, the poorest performance will probably be used as a basis for planning military missions, in order to avoid the possibility of losing airplanes through running out of fuel before returning to base. Thus, objectives that might be reached will not be attempted until, by much cut-and-try flying over a period of time, it is gradually determined that the mission can be increased in length.

This discouraging confusion in flight testing is now no longer necessary. We now have a correct analytical technique for handling our flight-test data. With this technique, we can obtain the basic lift and drag and thrust power from which we can construct charts to tell the pilot exactly how fast and far he can go as long as he maintains "proper" density altitude, power output, propeller r.p.m., weight and wind. These enable him to figure out quickly, for instance, in case he has had one engine shot out, how much power he can take from his other engines and still reach home with his remaining fuel. He can find out in advance exactly what his performance will be with any possible combination of weight and power because the basic equations of the airplane are known—something that can never be known from a flight test without this analysis. Furthermore, it is only by such analysis that it is possible accurately to predict all of the almost infinite number of combinations necessary to define exactly what the airplane will do at any altitude, any weight, any power, any r.p.m. of the propellers, any temperature, any wind, or any of the other variables desired.

Purpose of Performance Tests

The purpose of conducting flight tests on performance is normally to establish compliance with performance guarantees and to determine performance information necessary to permit intelligent use of the airplane. The increasing complexity of modern aircraft is forcing recognition of the fact that airplane performance can be improved not only by design but by determining the most efficient method of operation.

The modern transport and bomber airplane present complex performance problems inasmuch as they normally operate over a weight variation of as much as 2 to 1 and an altitude variation giving an atmospheric density ratio sometimes as great as 4 to 1. These airplanes are normally multiengined, and performance must be determined on various combinations of engines in the event of one or more power-plant failures. It is obvious that an infinite number of combinations of conditions exist and that flight testing would never end if all were checked individually. It has therefore

been necessary to develop a flight-test technique by which, with a limited amount of flight testing, performance under all conditions can be accurately predicted.

The basis for such a reduction in testing can be obtained by the use of performance estimation theory in conjunction with dimensional analysis. Following this scheme, a theory must be developed to explain all of the important effects obtained. The theory must be used to determine what flight conditions will separate the various effects and to insure that none is neglected. Flight tests must then be conducted in the average and extreme conditions to determine the exact effects of each variable.

Much of the basic theory of dimensional analysis as applied to flight testing was initially developed by A. C. Reed and Francis Clauser and is given in a paper by Mr. Reed.[1] In this paper, Mr. Reed outlined in detail a rigorous method of analysis of flight-test climb and level-speed data and explained the concepts of P_{tw}, C_{tw}, N_{tw}, and V_{tw}: power, climb rate, propeller r.p.m. and velocity, all reduced to standard weight and altitude. This method of handling flight-test data offered a solution at once to the controversial problems that have for 25 years kept our Army and Navy and the British from being able to compare airplane performances done by different services because each had its favorite method of reduction to standard. By examining the manner in which this method is applied to a few specific problems, we can begin to see the progress that has been made in "pinning down" to definite measurements many of the qualities of an airplane that were previously elusive because methods did not exist for isolating one effect from another and measuring it.

For instance, in the early days of flight testing, it was necessary to estimate propeller and engine performance, not separately but in combination with each other, on the basis of ground calibrations of engine and propeller. Propellers then were either fixed-pitch or two-position propellers, and no troquemeters were built to measure the horsepower. The modern airplane uses a propeller that is infinitely variable in pitch and on which the pitch is normally not recorded. The propeller is therefore of no use as a device for calibrating the engine in determining the engine power. Fortunately, torquemeters are now available for flight testing which permit an accurate evaluation of the power being absorbed by the propeller. A torquemeter is of primary importance in that it permits a complete separation of the aerodynamic performance problem from the power-plant performance problem and the performance of each may be studied entirely separately from the other in general. Once the engine has been separated from the performance problems, it becomes necessary to evaluate the aerodynamic performance of the propeller-airplane combination. With a fixed-pitch propeller, this is relatively simple since in this case a unique relationship of angle of attack to propeller advance ratio exists.

With the continuously variable-pitch propeller, it becomes necessary essentially to separate propeller performance from the airplane performance. This is not a simple problem, inasmuch as the only means now available for separating them is by the use of the airplane itself as a thrustmeter. This is somewhat complicated by unpredictable effects of propeller slipstream on the airplane drag. Furthermore, the "propeller-off" airplane drag characteristics are virtually never known and must, in general, be determined from a knowledge of the propeller thrust characteristics. It is obvious that a certain amount of cut-and-try work is involved in determining either the airplane drag characteristics or the propeller thrust characteristics.

A great deal of experience has been gained with airplane characteristics over a wide range of Reynolds numbers. There is appreciable change in drag with Reynolds number in many wind-tunnel tests, but there is little proof of its effect in flight. Its variation over the flight range of a given airplane for each angle of attack is generally negligible and, in general, it is safe to assume that the airplane drag characteristics do not change appreciably with Reynolds number. At the present time there has been little experience with the effects of high Mach number on the airplane drag characteristics, although considerable wind-tunnel information is now available on this subject. The propeller is not nearly so well understood. Propeller tips are always working at relatively high Mach numbers and invariably some conditions of flight involve the use of a propeller tip Mach number of the order of 1.0 where there is little doubt that major compressibility effects are taking place. A certain amount of wind-tunnel testing has been done to determine propeller characteristics, but all such testing has been performed at low speeds where Mach number effects would be negligible. Also only a limited number of propeller forms have been tested and no adequate means now exist for accurately predicting the performance of propellers as affected by changes in blade area, twist, airfoil section, etc.

The result of the above conditions is to make it necessary to test the airplane-propeller combination under widely varied propeller characteristics with constant airplane characteristics and vice versa. Several means are available for doing this. Tests can be conducted in level flight, climbs, and glides. These tests can be at high and low weights and high and low altitudes. Tests can be conducted in summer and in winter to give large variations in temperature. With a modern power plant and propellers, it is possible to make tests with a given input power at both high and low r.p.m.'s. By use of dimensional analysis, a series of tests can be planned with sufficient variation in variables so as to determine the effects of the variables on the performance of the airplane. The best way to

explain the use of this procedure of flight testing is to describe the method by which the data are reduced.

P_{tw}-V_{tw}-N_{tw} Method of Reduction of Level Flight Data

In the case where compressibility effects are negligible and for first approximations, the V_{tw}-P_{tw}-N_{tw} method given in Reed's paper may be used where V_{tw}, P_{tw}, and N_{tw} are velocity, power, and propeller r.p.m., modified for variations from standard air density and gross weight.

$$V_{tw} = \sigma^{1/2} V_{\text{true}}/(W/W_{\text{std.}})^{1/2}$$

$$P_{tw} = \sigma^{1/2} \text{ b.h.p.}/(W/W_{\text{std.}})^{3/2}$$

$$N_{tw} = \sigma^{1/2} \text{ r.p.m.}_{\text{propeller}}/(W/W_{\text{std.}})^{1/2}$$

This method is completely rigorous for cases where Reynolds number and Mach number effects are negligible. In general, Reynolds number effects can be considered negligible. This is not normally true of Mach number effects. The V_{tw}-P_{tw}-N_{tw} method is an accurate method of presenting flight data without distortion. In order to obtain data for presentation by this method, it is best to fly the airplane at several different speed-power conditions but with the engine set at constant r.p.m. From such test data, plots of P_{tw} vs. V_{tw} for constant values of N_{tw} can readily be constructed. Two such plots are presented on Figs. 1 and 2, Fig. 1 representing tests run with an airplane equipped with a propeller of average solidity and Fig. 2 representing tests run with a propeller of high solidity. This method of presentation is not, however, a convenient method of predicting performance and for interpolation. For this reason it is common practice

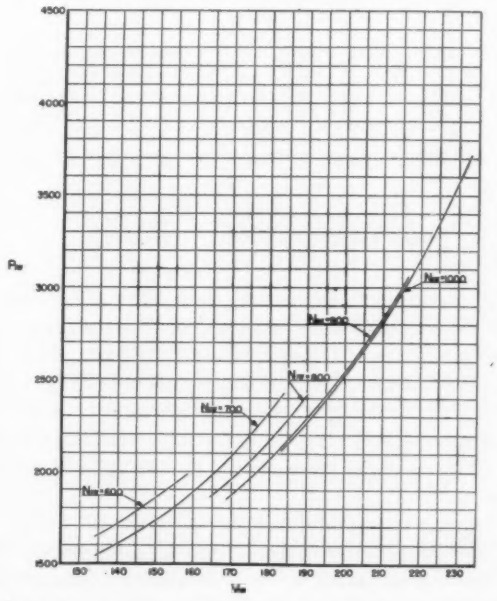

FIG. 1. P_{tw}-V_{tw}-N_{tw} curves for airplane with propeller of average solidity.

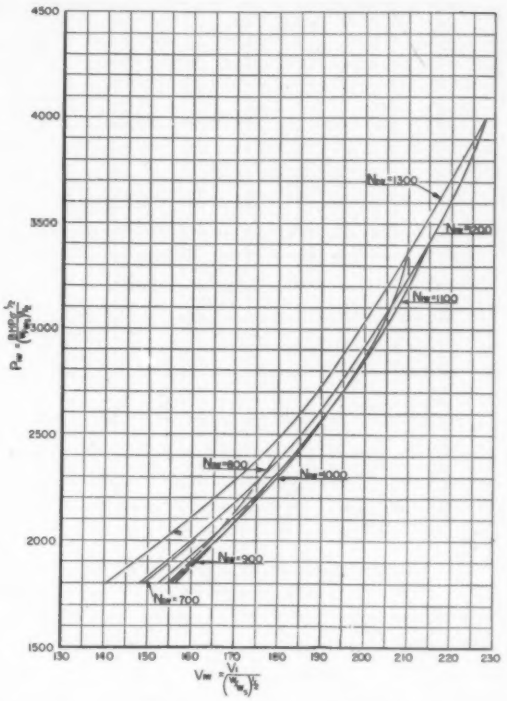

FIG. 2. P_{tw}-V_{tw}-N_{tw} curves for airplane with propeller of high solidity.

to reduce the $V_{tw}P_{tw}N_{tw}$ data to an airplane polar and equivalent propeller chart. In doing this, few assumptions are required, and these can be completely verified if the results of the polar and equivalent propeller chart are used to compute P_{tw}-V_{tw}-N_{tw} curves identical to those obtained in flight.

Airplane Polar Method of Reduction of Level Flight Data

An equivalent airplane polar and propeller chart may be determined by taking a chart for a propeller which may be the same as, or similar to, that incorporated on the airplane and estimating a propeller efficiency. The values of drag coefficients, where $C_D = D/qS$, and lift coefficient, where $C_L = L/qS$, are then computed for each point and plotted as C_L^2 vs. C_D. If all points fall on a line, it may be assumed that the propeller is performing according to the chart. This will not normally be the case if a sufficiently wide variation of propeller loadings were used in a flight test. A convenient way to modify the propeller chart for differences between an assumed and an actual propeller is to multiply the power coefficients by a constant. This is particularly true when using a propeller chart in the form of efficiency, η vs. advance ratio, $J = 88V/ND$, at constant values of power coefficient, C_P. Examples of this type of analysis are shown in Figs. 3a, 3b, and 3c. Fig. 3a shows test points reduced to C_L^2 vs. C_D form where the test propeller was of higher solidity than the chart propellers, and no correction was applied to C_P in the reduction of the data.

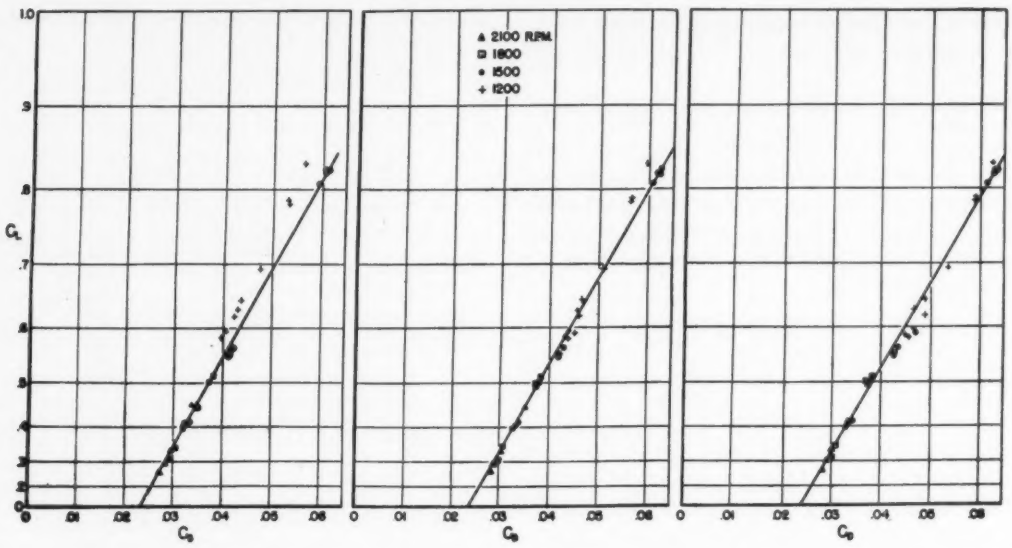

FIG. 3a. Airplane polar—data reduced with no propeller solidity correction. FIG. 3b. Airplane polar—data reduced with suitable solidity correction. FIG. 3c. Airplane polar—data reduced with too great a solidity correction.

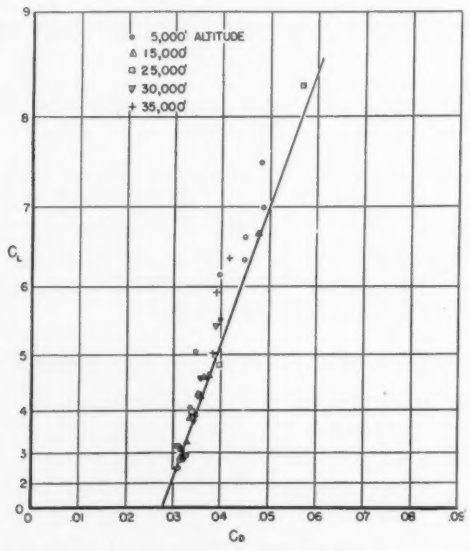

FIG. 4a. Airplane polar for high solidity propeller—no solidity correction applied to propeller data.

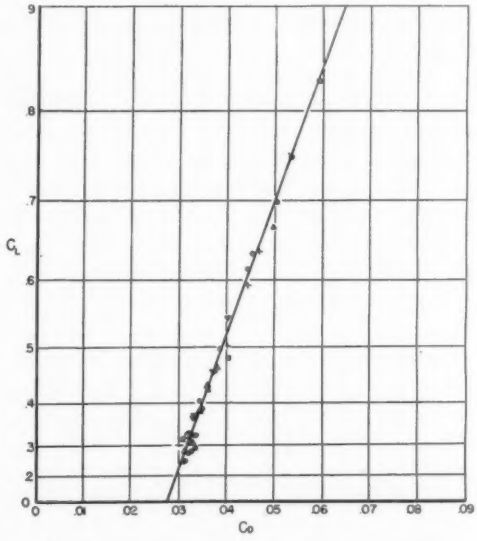

FIG. 4b. Airplane polar for high solidity propeller—suitable solidity correction applied to propeller data.

Fig. 3b shows the same data reduced to polar form, although in this case approximately the correct solidity correction has been applied to C_P in the reduction of the data. Fig. 3c shows the same basic data, but in the reduction of these data too large a solidity correction has been applied to the power coefficient. A more rational method of reduction requires the multiplying of C_P by a constant and J by the cube root of the constant. By assuming several different values of the constant and reducing the data to C_D vs. C_L^2 for each point, a value may be determined which most nearly makes the data fall in a continuous line and gives the least scatter of points. In general, it has been found that the proper constant for correcting C_P is close to the ratio of the activity factor of chart blades to the activity factor of the test blades. An example of data reduced by the latter method is shown in Figs. 4a and 4b. Fig. 4a is plotted with no solidity correction applied and Fig. 4b is plotted with a solidity correction applied which is proportional to the ratio of activity factors.

Frequently, it will not be possible to remove all scatter of points by the choice of a constant. This remaining scatter may be caused by Mach number effects on either the airplane or the propeller. On most present-day airplanes, any effect of Mach number

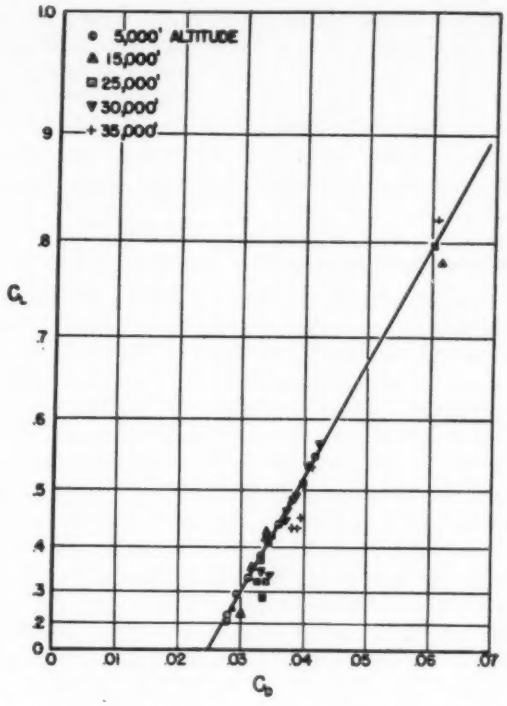

FIG. 5a. Airplane polar—no propeller tip speed correction applied to data.

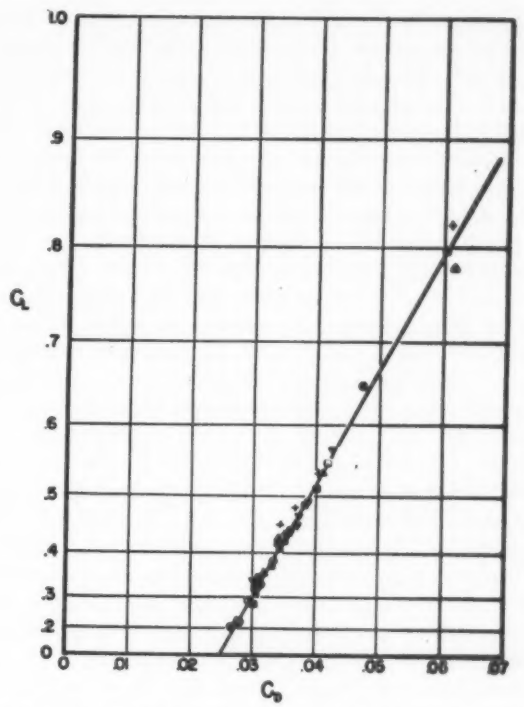

FIG. 5b. Airplane polar—suitable propeller tip speed correction applied to data.

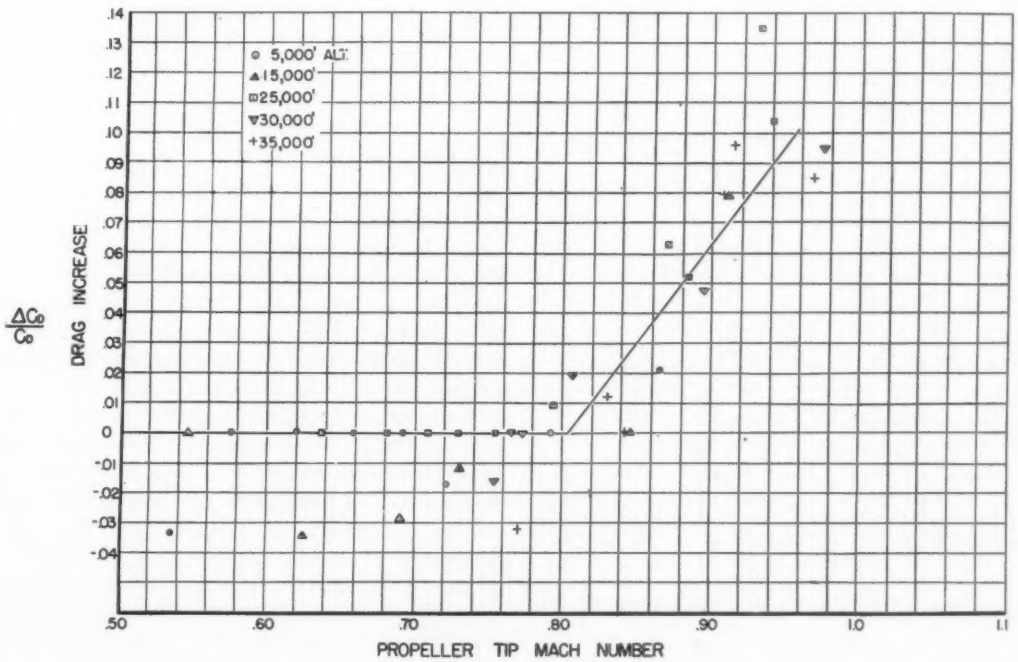

FIG. 6. Effect of high propeller tip speeds on airplane drag.

on drag may be assumed negligible, but the effects upon the propeller itself may be quite large. If the propeller tip Mach numbers are listed for each point plotted on the C_L^2 vs. C_D plot, any consistent variations with propeller tip Mach number may be readily noticed. An example where considerable scatter of the points was caused by high propeller tip Mach number is shown in Fig. 5a. These Mach number

effects may be evaluated by making a plot of percentage deviations in the drag direction from each point vs. propeller tip Mach number as shown in Fig. 6. It will normally be found that a consistent effect of propeller tip Mach number can be predicted in this manner. By the above method an equivalent propeller chart and propeller Mach number effect may be determined. The data shown in Fig. 5a, with a Mach number correction so determined applied in the reduction, is shown in Fig. 5b. These data may be checked by using the propeller chart and propeller Mach number chart to estimate propeller efficiency and then plotting THP_{tw} vs. V_{tw} for each flight-test point. The assumed drag curve can then be reduced to this same form by use of the equation:

$$THP_{tw} = V_{tw}^3 C_D/146{,}600$$

The resultant accuracy of the assumed propeller and airplane characteristics can thus immediately be evaluated in terms of horsepower and speed. If the scatter is of an acceptable magnitude, the results may be considered satisfactory. An example of such correlation is shown in Fig. 7. The flight-test points are the same points as shown on the polar in Fig. 5b. The parabolic curve drawn through the points represents the straight line variation of C_D with C_L^2, also shown in Fig. 5b.

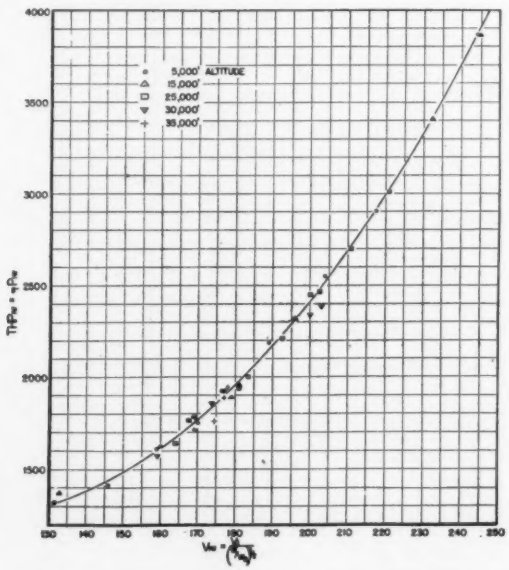

FIG. 7. THP_{tw}-V_{tw} curve for data shown in Fig. 5b.

These results are considered satisfactory and it is thus possible, by the use of such data, to obtain charts to determine the aerodynamic performance of the airplane under any assumed conditions using the drag polar and propeller charts.

In general, a determination of the propeller chart and Mach number chart need be done only for the standard configuration of the airplane. This propeller chart may then be used in the reduction of flight data concerning the effects of various items of drag such as cowl flaps, wing flaps, and landing gear. The use of the propeller charts directly in obtaining partial engine performance must be effected with due caution. Complete test data for these conditions must be obtained.

Reduction of Climb Data

Reduction of climb test data is usually quite difficult. Since climbs are continuous in most cases, certain portions of data will invariably be obtained under weather conditions not conducive to accurate testing. Furthermore, most airplanes are not particularly stable longitudinally in rated power climbs and inherently have a change in longitudinal trim with change in altitude at rated power. For these and other reasons, the speed of the airplane cannot normally be held within several m.p.h. of a mean value in a climb. In analyzing individual test points, it is necessary to correct for speed fluctuations from the mean. A first approximation of such a correction can be made by equating the differentials with respect to speed and altitude of the kinetic and potential energies of the airplane.

$$\begin{aligned} KE &= {}^1/_2(W/g)V^2 \\ PE &= Wh \\ dKE &= (W/g)VdV \qquad dPE = Wdh \\ \text{assume } dPE &= -dKE \\ (w/g)VdV &= -Wdh \\ dh &= -(V/g)dV \end{aligned}$$

At a climbing speed of 150 m.p.h., this correction is 10 ft. for each 1 m.p.h. These speed fluctuations are not measurable unless data are frequently obtained. A satisfactory method for doing this is to obtain simultaneous photographic recordings of air speed, altitude and time every 10 sec. Another correction that must be applied to the climb data in determining rate of climb is that for the change in actual altitude rather than pressure altitude. Inasmuch as differences in altitude are obtained from the readings of an altimeter, a difference in pressure is measured such that the true altitude change is indicated only when the air temperatures are the same as those that would exist in the "standard atmosphere" at the "pressure" altitudes. It is therefore necessary to apply a temperature correction to obtain the actual change in altitude. The altimeter is calibrated so that ΔP, which the altimeter measures, is equal to Δh_i (height change shown on altimeter) $\times$ $\rho_{\text{std.}}$. When the air temperature is not standard, $\Delta P = \Delta h_{\text{actual}} \times \rho_{\text{actual}} = \Delta h_i \times \rho_{\text{std.}}$. Thus

$$\Delta h_{\text{actual}} = (\Delta h_i)(T_{\text{actual}}/T_S)$$

The actual rate of climb during a given time interval is thus equal to the observed rate of climb $\times$ T_{actual}/T_S.

Even though the above energy and temperature corrections are applied, the data taken at frequent time intervals may show wide scatter if plotted in the form of rate of climb vs. altitude. For this reason it may be necessary to average the rates of climb over intervals of from 500 to 1,000 ft.

Power-Plant Performance Tests

Fortunately, the torquemeter presents an accurate method of determining the power output of the engine for any given test point, and thus a major variable is eliminated during performance testing To be able to predict the performance of the airplane, however, it is necessary to know how the engine will perform over as wide a range of temperature and pressure conditions as the airplane will be required to operate. Usually, simultaneous recordings of all essential engine performance measurements can be made at the same time the performance recordings are made and, therefore, a great deal of the power-plant performance data can be obtained during the airplane performance testing. The data that must be determined fall into two categories: (1) Power-Plant Operating Characteristics at All Altitudes; (2) Fuel Consumption Characteristics at All Power Conditions.

Power-Plant Operating Characteristics

Meticulous and complete information on the operation of the airplane power plant under all conditions is important because of the complexity of the modern aircraft power plant and because of the extreme amounts of power now being derived from it as a result of constant improvement. This field of testing has been greatly developed recently. The amount of test work necessary will depend largely upon the installation. If the engine is equipped with a mechanically driven supercharger, the testing will consist of determining the engine critical altitudes and the variation of full throttle power with altitude. In addition, engine temperatures, pressure drop through the engine, and cowl flap settings for the required pressure drops must be determined. The reduction of these data to standard conditions has been adequately covered by D. S. Hersey.[3]

The problem of determining engine performance becomes considerably more complex when the installation is equipped with a turbosupercharger. The speed criticals of the turbo—that is, the altitude at which the turbo wheel reaches the maximum r.p.m. ratings—must be measured as well as the limits of instability and closed waste gate. These latter test conditions will occur at less than rated power and are dependent to a large extent on the design of the diffuser of the turbocompressor and on the design of the nozzle box. Testing and properly evaluating the complications of such a system have been made possible by the use of elaborate instrumentation on the test airplane. The development of instruments to measure individually the many factors of power-plant performance which formerly were measured only as a group has been one of the important advances in present-day flight testing. Especially has this been feasible in large four-engined aircraft, in which the size of the craft makes possible the carrying of a considerable array of test equipment and a sufficient test crew to operate it. In general, the following instrumentation will suffice as a minimum: (1) engine r.p.m.; (2) engine b.m.e.p. (torquemeter); (3) engine manifold pressure; (4) turbo r.p.m.; (5) carburetor altitude; (6) exhaust back pressure; (7) static pressures aft of oil cooler, intercooler and engine baffles; (8) total pressures ahead of engine; (9) cowl flap, intercooler flap, oil cooler flap, and waste gate position indicators; (10) carburetor inlet temperature; (11) cylinder head and base temperatures; (12) outside air temperature; (13) air speed; (14) altitude; (15) exhaust gas temperature; (16) fuel consumption.

By taking simultaneous test data with all of the above instrumentation during tests that should cover the range of operating conditions to which the engine will be subjected, an engine calibration independent of the airplane performance can be obtained. It is desirable that the more critical conditions of the engine operation be checked at various altitudes in both summer and winter so that as wide a range of temperatures as possible may be covered. These tests may then be reduced to the form of an engine calibration of b.h.p. available versus altitude for various operating r.p.m.'s. In this manner, the power output of the engine under any condition is shown—independent of the airplane performance.

Since the range of an airplane is a primary item of performance, the fuel consumption characteristics of the engine must also be determined. For these tests, it is desirable to determine the specific fuel consumption with automatic mixture settings as well as manual settings. Such tests should cover the fuel consumption characteristics of all engines on multiengined ships if the installations differ in order to determine whether any carburetor air scoop effects exist. Since scoop effects and variations in fuel consumption with manual leaning may be small, an accurate measuring device is necessary. In general, the majority of commercial-type fuel flowmeters are not satisfactory for precision flight tests of fuel consumption. A volume-type flowmeter can be constructed, however, with which the time for a given volume of fuel to be used can be measured with a stop watch. Exceptional accuracy is possible with such a flowmeter. Examples of fuel consumption data obtained with such a setup are shown in Fig. 8.

By using these data in conjunction with the engine calibration, the airplane polar, and propeller chart, the range of the airplane under any combination of possible operating conditions can be readily predicted. In fact it is now common practice to construct a chart

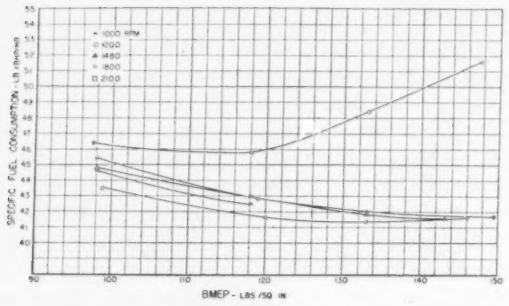

FIG. 8. Flight-test fuel consumption data with automatic mixture control setting.

from these basic data showing the pilot how he can get the maximum range for any weight or altitude and, in addition, what the range will be for any fuel load, power, weight, altitude and temperature that is likely to be encountered. These charts have proved highly valuable in military and long-distance ferrying operations and provide a splendid example of the benefits of scientific flight testing.

Testing Essentials for Obtaining Satisfactory Performance Data

The purpose of performance testing and methods of reduction of the data to usable forms have been considered; to obtain data satisfactory for such use, however, the technique of testing must be carefully controlled. This technique required for performance testing covers careful selection of the crew of the airplane, accurate instrumentation photographically recorded during flight, accurate laboratory and flight calibrations, careful control of the weight and configuration of the airplane, and caution as to the weather conditions under which testing is conducted. If proper consideration is not given to minimizing the errors that may be introduced from such sources, the obtaining of data satisfactory for the type of analysis outlined will be impossible.

Crew Considerations

Much of the progress of recent years in flight testing has been made possible by taking a fully trained and integrated engineering research crew aboard on test flights where formerly it was the job of a single test pilot. In large multiengined airplanes, a flight-test crew usually will consist of from five to ten men. Obviously, unless all of these men are experienced in their duties, the testing will not be entirely successful. Assuming, however, that a majority of the necessary data can be recorded photographically, the crew may be reduced to a minimum of three. The most important from the standpoint of obtaining performance data are pilot, copilot, flight engineer, and aerodynamic observer.

It is assumed, of course, that the pilot and copilot are competent; however, in addition to this, they must be sympathetic toward the type of testing that is being done, i.e., willing to hold the airplane exactly at one condition even though routine and extremely tiresome. It is desirable that they have, in addition, an understanding of the fundamentals of airplane performance and what is to be done with the data.

The performance or aerodynamic observer should be a man who has spent considerable time on analysis of the subject test item and also should be the one who will reduce and analyze the data obtained. It has been found that if such an engineer is present during the testing a more intelligent analysis of the tests can be made. Also, such an observer who is able to anticipate results will know exactly how successfully the test is progressing, in addition to being acquainted with the circumstances under which the data were obtained. The data may be reduced and plotted in flight during the test, thus facilitating the recognition of poor test points and the prevailing limits for testing. In case of unusual data, he may offer advice as to direction of the testing in order to obtain the most valuable information for his needs. This does not mean that the test should not be carefully planned beforehand but means that in case of unpredicted results sufficient changes in plans can be made to obtain complete and satisfactory data.

Instrumentation

The importance of accurate instrumentation cannot be overstressed. Most of the instruments listed above in the section, "Power-Plant Operating Characteristics," are essential for engine performance. Laboratory calibration of all instruments during the test program reveals those instruments that are too inaccurate or do not hold their calibration.

Air-Speed Measurement

An accurate laboratory calibration is not enough for the air-speed indicator, inasmuch as this most important instrument has an independent calibration depending on the angle of attack. This calibration may also be readily affected by leaks in the pitot static system. For this reason it has been found essential that the total and static lines be thoroughly checked for leaks immediately before and after each performance flight. Difficulties that may be encountered by not observing this precaution are shown by the curves in Fig. 9.

The test data shown in this example have been reduced to the form of the ratio of true indicated air speed, V_t, to the actual indicated air speed as registered through the pitot installation, V_I. This ratio is plotted against the parameter

$$V_I/(W/W_{\text{std.}})^{1/2}$$

where

V_I = actual indicated air speed as registered through the pitot installation, independent of instrument error

W = weight at time of test
$W_{std.}$ = arbitrary weight chosen as the standard weight of the airplane

For a given configuration, there will then be only one value of V_t/V_I for any one value of V_{Iw}.

The curves shown represent four different calibrations obtained on the same airplane with the pitot masts in exactly the same location. The only change in the airplane was a refinement in the surface just forward of the pitot masts. This change was made between the running of calibrations 1 and 2.

Calibration 1 was obtained early during the testing of the airplane. This calibration was later checked and used for a series of performance tests. Calibration 2 was run after the surface ahead of the pitot mast had been improved. At the time it was assumed that this change in the calibration was due to the modification of the airplane. Calibration 3 was obtained about two months later and prior to running another series of performance tests. The wide variation between calibrations 2 and 3 was not explainable at the time inasmuch as there were no apparent leaks in the system. Nevertheless, when a recheck was run, calibration 4 was obtained. The pitot system was again inspected and it was found that there was a slight leak not apparent under the ordinary pressures used for normal checking but possibly of some magnitude when in flight and subjected to vibration. A rerun of the calibration checked the previously obtained calibration 4 which, it is to be noted, closely checks the original calibration 1. The reason for the difference in calibration 3 may, of course, be explained by a leak in the system. The reason for calibration 2 appearing as it did was not explained satisfactorily, however, unless the variation was due to flying the airplane at a slight angle of yaw during the calibration runs.

Thus, it can be seen that even though care may be taken in the selection and calibration of the air-speed indicator, the measured speed may be several per cent off because of an unreliable air-speed calibration. The only method that has been evolved to cope with this situation is to check carefully the pitot system for leaks immediately before and after each flight, in addition to checking the air-speed calibration by several runs over a speed course prior to an extensive series of performance tests. Unless the pitot static of the airplane has been proved to be relatively insensitive to angles of yaw, the accuracy of the test will be quite largely affected by flying either the speed course calibration or the speed runs at an angle of yaw.

Outside Air Temperature Measurement (Free Air Temperature)

The measurement of outside air temperature, while apparently a simple procedure, is really quite difficult if it is to be done accurately. The type of element,

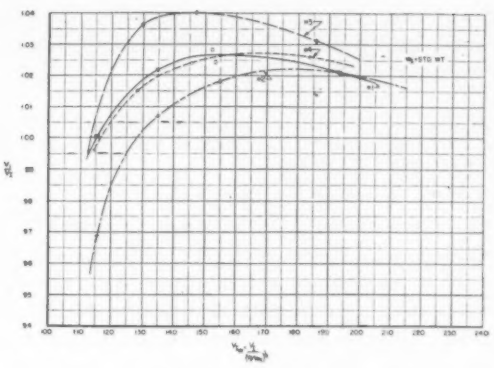

FIG. 9. Air-speed calibration variations obtained with same airplane.

location of the element, and method of recording the temperature are all items that must be carefully considered if accurate performance testing is to be done. In addition, each element will have a correction to be applied due to the temperature reading being high from the effects of adiabatic compression temperature rise caused by the compression of the air in front of the temperature measuring element, and boundary layer friction of the air. Flight tests have proved this correction to be between 0.80 and 1.0 times the full adiabatic temperature rise and, consequently, a function of true air speed and not of density. The adiabatic temperature rise is expressed by the equation

$$T_{ad} = JV^2/2gC_P$$

where

J = mechanical equivalent of heat
V = true air speed in ft. per sec.
g = is the acceleration due to gravity (32.17 ft. per sec.^{2})
C_P = the specific heat at constant pressure

This can be more conveniently expressed by the equation

$$T_{ad} = \left(\frac{C_P/C_V - 1}{C_P/C_V}\right)\left(\frac{V^2}{2gR}\right)$$

where C_P/C_v equals 1.4 and is the ratio of the specific heat at constant pressure to the specific heat at constant volume. R is the universal gas constant (53.33). If V is in m.p.h. and degrees centigrade is being measured, the true adiabatic rise is then

$$T = V_{true}^2 \times 9.94 \times 10^{-5}$$

and for degrees Fahrenheit

$$T = V_{true}^2 \times 1.79 \times 10^{-4}$$

At 0.80 of the full adiabatic temperature rise, this correction is of the order of 10°C. at 350 m.p.h. true air speed. The importance of considering this variable should therefore be readily appreciated. A satisfactory method of checking the constant for the particular

element in use is to run a series of speed runs at both low and high altitude and thereby check the temperature rise over a wide range of speeds.

Torquemeters, Tachometer, Manifold Pressure, Carburetor Air Temperature, Carburetor Deck Pressure, and Exhaust Back Pressure

All of the above instrumentation is used for determining the power output of the engine. Ordinarily the torquemeter and tachometer are sufficient for this; however, it is impossible to dispense with the other measurements. Torquemeter readings have quite frequently been found to be erratic, and for this reason, unless careful cross checks of chart power against torquemeter power are made, misleading results may be obtained.

Control of Weight and Configuration

Weight. The precision of flight test data is exactly proportional to the precision with which the weight of the airplane is determined. A whole series of test results may be rendered invalid by inaccuracies in weight calculation. The most satisfactory method of weight control would be to weigh the complete airplane immediately before and after each flight. Inasmuch as this is not ordinarily possible, computed weights are usually relied upon. This method ordinarily is satisfactory if a capable man is in charge of the weight control of the airplane. In addition to the above, frequent fuel quantity readings must be taken during flights in order to establish the weight at any time during the flight. Failure to observe this precaution may result in the inability to determine the weight within 3 per cent of the actual condition.

Configuration. This item concerns control of the outside arrangement of the airplane, lest an unintentional change in drag should cause a variation in results which might be falsely attributed to other factors. Here two conditions are important: One is observance of the rule that absolutely no external items be added to the airplane during performance tests; the other rule is that all items such as cowl flaps, intercooler flaps, wing flaps, landing gear, bomb doors, etc., be carefully checked immediately prior to any performance testing. Although this item is apparently simple to control, it has been found on experimental airplanes that frequently one or more of these automatically closing devices is enough out of adjustment to give appreciable differences in diagnosis.

Data Recording

The modern multiengined ship, because it is a labyrinth of interrelated mechanisms, has brought forth a multitude of problems in the way of data recording during flight tests. It is highly desirable, in fact almost essential, that simultaneous readings of power conditions, air speed, and altitude be taken if satisfactory data are to be obtained. Thus, in order to obtain the type of information desired it is necessary to use automatic recording devices. Today, there are excellent temperature recorders available and, where space allows, it is no great problem to install a photographic recorder containing all essential engine and flight instruments. With such an arrangement, simultaneous photographic recordings of all instruments can be obtained as often as desired.

The value of such information is illustrated in Fig. 10, Runs 1, 2, and 3. In these illustrations, photographic recordings of time, altitude, and air speed were taken every 2 sec. In each case the purpose of the run was to obtain a stabilized level speed performance run. On Run 1 both the altitude and air speed varied sufficiently to leave the result in doubt, although at the time the test was run it was considered satisfactory.

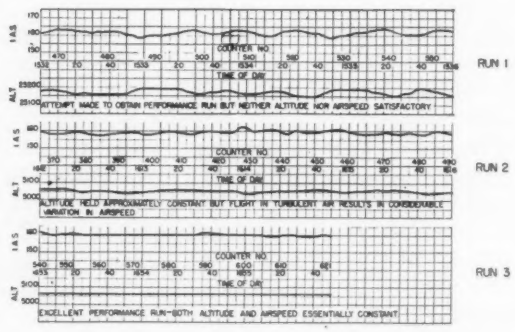
IAS
ALT
COUNTER NO.
TIME OF DAY
ATTEMPT MADE TO OBTAIN PERFORMANCE RUN BUT NEITHER ALTITUDE NOR AIRSPEED SATISFACTORY
RUN 1
ALTITUDE HELD APPROXIMATELY CONSTANT BUT FLIGHT IN TURBULENT AIR RESULTS IN CONSIDERABLE VARIATION IN AIRSPEED
RUN 2
EXCELLENT PERFORMANCE RUN—BOTH ALTITUDE AND AIRSPEED ESSENTIALLY CONSTANT
RUN 3

FIG. 10. Air-speed-altitude relationships measured during performance tests.

Run 2 is an example where the pilot did an excellent job of holding the altitude constant, but the speed was somewhat erratic owing to slight gusts. This test is considered satisfactory, although it would be difficult to determine the exact speed without the photographic recording.

Run 3 is an example of excellent piloting. Here both speed and altitude are so nearly constant that there is no doubt of the answer.

The value of these photographic recordings is that: (1) the flight can be analyzed and judged on the ground after landing when time is not the limiting element; and (2) a permanent record is obtained of the test such that any questionable points can be checked. An additional advantage of using a photographic recorder is that it enables a trained observer to watch all of the pertinent instruments from a vantage point heretofore impossible. Tests may be thus roughly analyzed in flight and the success of the testing can be determined as the flight progresses. Two photographs showing the camera and instrument panel in a modern photorecorder are given in Figs. 11a and 11b.

Weather Conditions

The weather is one major variable in flight testing which man unfortunately cannot control. Inas-

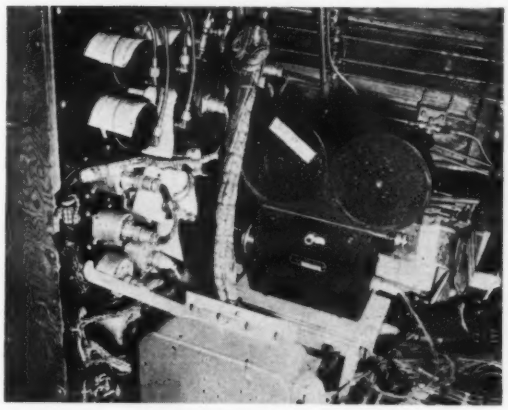

FIG. 11a. Modern photorecorder installations.

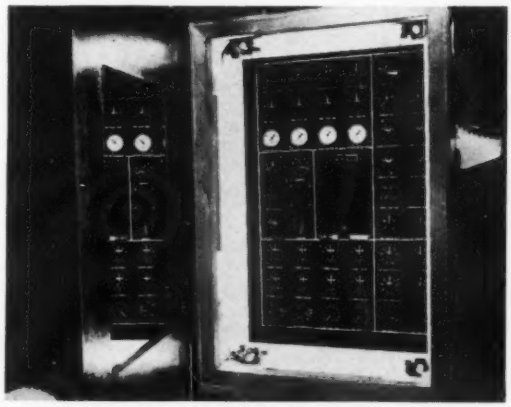

FIG. 11b. Modern photorecorder installations.

much as this is so, the rule then must be that performance tests are run only in favorable weather conditions. If the rule is to be followed, the problem arises as to just what are satisfactory conditions. Flying in clouds, near clouds, or in rough or turbulent air will, in general, produce such erratic results that data obtained under such conditions will be no better than a rough approximation of the answer. In general, clear weather, free of clouds, will be satisfactory for testing; here again, however, precautions must be observed or the testing will result in misleading answers. It has been found that the lapse rate, i.e., the rate of change of the outside air temperature with change of altitude, is of definite significance in determining satisfactory or unsatisfactory conditions. Recent data have shown that if tests are performed in an area where the lapse rate departs appreciably from normal in either a stable or unstable direction the tests are likely to be unsatisfactory. In general, it may be said that if the lapse rate becomes greater than the dry adiabatic or less than the wet adiabatic at any time the conditions will be unsatisfactory near the point of change in lapse rate.

Several examples of what may happen when changes

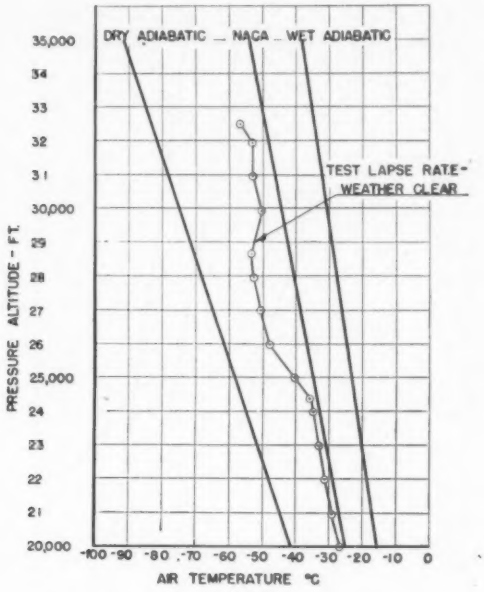

FIG. 12a. Lapse rate during climb test.

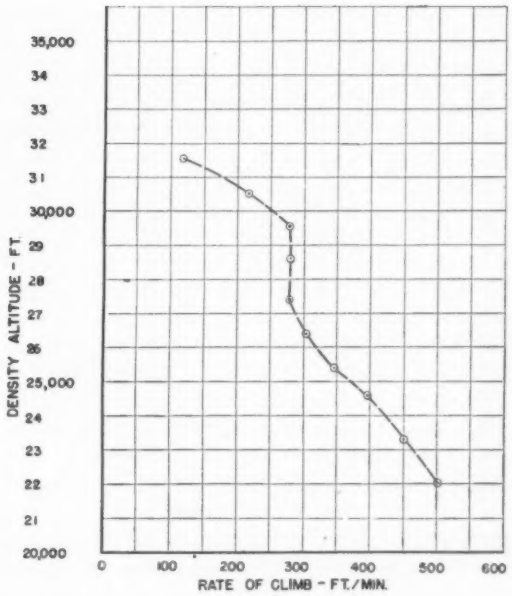

FIG. 12b. Rate of climb during test with temperature lapse rate in Fig. 12a.

in lapse rate are encountered are shown in Figs. 12, 13, 14, and 15. In Fig. 12, a curve of rate of climb versus density altitude is presented with the corresponding lapse rate during the test. Here it can be seen that the inconsistency in the rate-of-climb curve corresponds closely to the changes in lapse rate during the test. Fig. 13 is a similar comparison, only in this case the end of the troposphere has been reached and there is no further decrease in temperature with altitude. Apparently this effect is also responsible

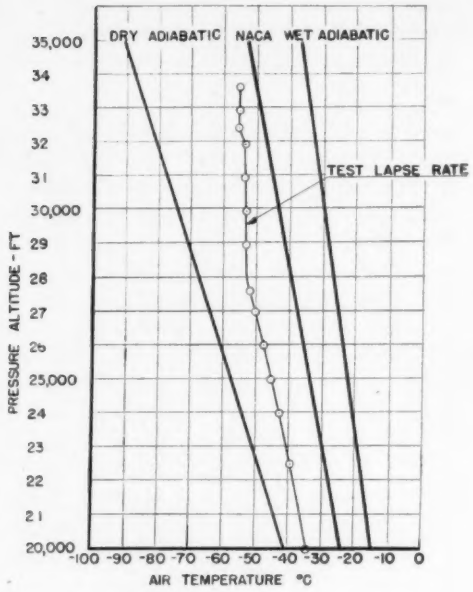

FIG. 13a. Lapse rate during climb test.

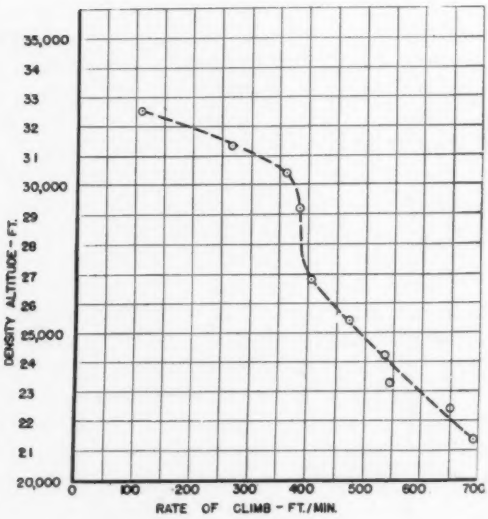

FIG. 13b. Rate of climb during test with temperature lapse rate in Fig. 13a.

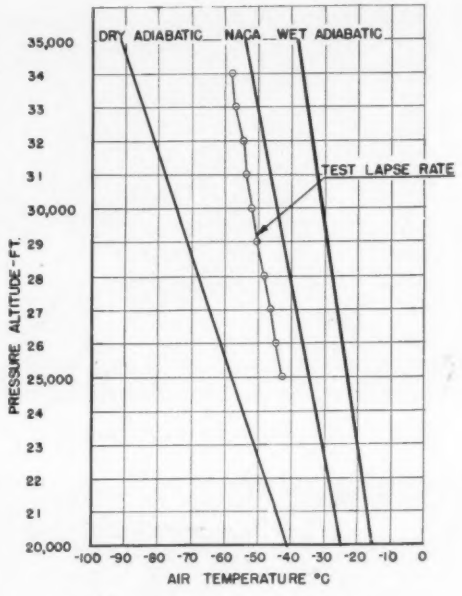

FIG. 14a. Lapse rate during climb test.

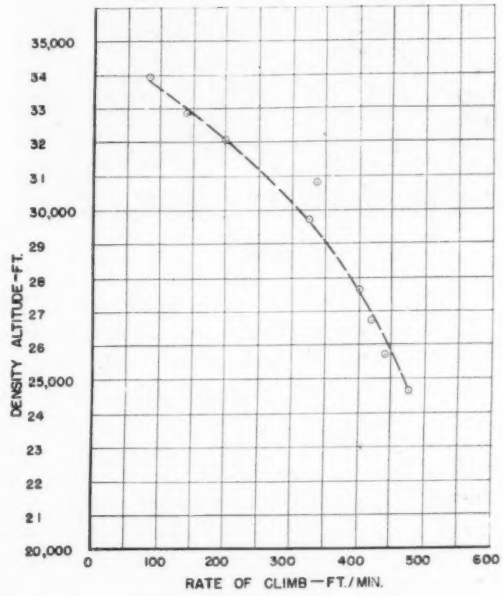

FIG. 14b. Rate of climb during test with temperature lapse rate in Fig. 14a.

for the unusual shape of the rate-of-climb curve shown. Fig. 14 is another similar example; in this case, however, the lapse rate was quite normal throughout the test. The test data here show in general a quite consistent variation in rate of climb with increase in density altitude.

On all of the above tests the weather was perfectly clear in the test region. More information on the effect of lapse rates on testing is shown in Fig. 15a. In this case there was a definite inversion. The weather was hazy up to about 4,000 ft. A level speed performance run was attempted at 6,000 ft., and differences in stabilized air speed of 3 m.p.h. were measured while flying over the area chosen for testing.

It may be concluded then that testing is likely to be unsatisfactory whenever the lapse rate varies appreciably from normal. One additional precaution that should be noted, however, is that a normal lapse rate and clear weather do not always indicate satisfactory weather for performance tests. Fig. 15b illustrates a case where the observed lapse rate was perfectly normal; nevertheless the differences in stabilized

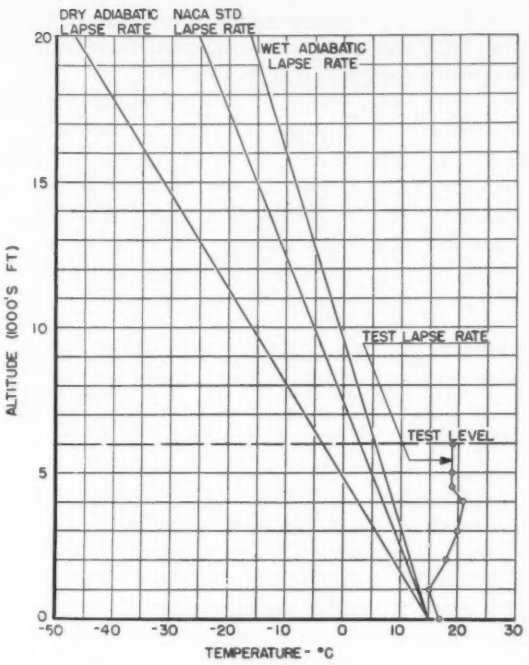

FIG. 15a. Temperature lapse rates measured prior to performance tests.

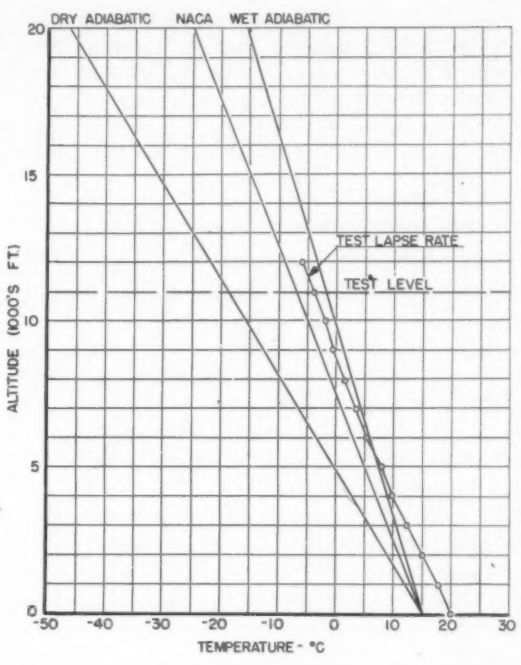

FIG. 15b. Temperature lapse rates measured prior to performance tests.

air speed measured while flying in various directions over the test area was 10 m.p.h. It was observed at the time that clouds were forming at approximately 40 miles distance at the altitude of flight. Apparently, in this case, there was a vertical wind gradient of sufficient magnitude to cause the discrepancies in air speed.

Briefly, then, if satisfactory performance tests at altitudes are to be made, the following rules should be followed concerning the weather:

(1) Do not fly near or among clouds.

(2) Determine the lapse rate over the test area either prior to the test or during the test by plotting temperature vs. altitude in a climb. Do not conduct tests where the lapse rate is changing or near changes in slope.

(3) Prior to running level speed performance runs, check the test area for adverse wind gradients by holding constant the altitude, r.p.m., and b.h.p. and noting any changes in air speed.

It is believed that if these rules are observed variations in performance tests due to weather can be virtually eliminated.

In all factors influencing performance testing, whether it be the weather, the weight or configuration of the airplane, the condition of the power plant, or the effectiveness of the instrumentation, a mania for meticulous accuracy and a detective eye for all the conceivable variables are essential. Only by this attitude of mind have we been able to advance the art of test flying to an exact science that has, bit by bit, pushed back the margins of airplane performance.

STABILITY

While performance is often central in the flight-testing program, there is another respect in which the airplane must be tested—namely, with reference to its flying qualities. Testing for these qualities is fully as important as that for performance, and it is also no less quantitative and accurate. These flying qualities, stability and controllability, are of particular importance on large airplanes that must be flown many hours at a stretch. In such airplanes, poor stability and controllability will result in fatigue and inefficiency of the crew and decrease the effectiveness of the airplane no less than poor performance will decrease it. In addition, these features may prove the difference between success and failure in the tight spot of a military mission when all the airplane's flight characteristics come forth to its advantage or disadvantage.

In the testing of these flight characteristics, the test crew was at first confronted by the problem of developing not only a method of testing but also a system of measurement of such an abstract concept as a flying quality. Then came the problem of linking flight knowledge with design knowledge in order to translate desirable flight characteristics into new design configurations that would produce the desired results in the air. By this process, it has been possible gradually

to analyze and give aerodynamic design treatment to marginal flight conditions that once were either unknown or not understood. The problems of stability and controllability are so varied that no attempt can here be made to outline them completely. Certain problems only will be discussed to indicate the method of analysis.

Static Longitudinal Stability Tests

There are several variables, measurable in flight, which have a definite relationship to the static longitudinal stability of an airplane. Some of the more common of the stability criteria are: (1) rate of change of elevator force or hinge moment with air speed; (2) rate of change of elevator position with air speed; and (3) rate of change of airplane pitching moment with air speed. Measurement of elevator forces and positions in flight is relatively straightforward once the airplane is instrumented for such measurements. The first two methods of measuring stability are well known and will not be dwelt upon here. Direct measurement of the airplane pitching moment is, however, a more lengthy process because it necessitates the shifting of the airplane center of gravity in flight. This method, now established by flight-test procedures and aerodynamic analysis, is superior to former methods in that it reveals the complete longitudinal stability situation, whereas the other methods cannot be carried throughout the entire range of stability. In addition, they are limited in application to small airplanes or those whose control-system friction is low.

The most convenient of the stability criteria for analysis, and the most basic from the pilot's point of view is $-dC_M/dC_L$; therefore, the flight technique that measures this value in the most direct manner, both for free and fixed elevators, is the most desirable. It is possible to compute the fixed elevator pitching moments from elevator position curves if sufficient data have been taken so that the elevator effectiveness factor is obtained. A more direct procedure is the so-called "effective weight moment method." This procedure measures free, as well as fixed, elevator stability.

In brief, this method consists of measuring the change in trim speed associated with a change in c.g. position, while the other variables (power, altitude, and elevator trim tab setting) are held constant. A detailed discussion of the procedure followed and the flight test technique used is given below.

Airplane Trim Conditions

For the measurement of static longitudinal stability, it is necessary that the airplane be longitudinally trimmed ($C_M = 0$) when data are taken. In order that the flight test procedure discussed in the next section can be more easily followed, a discussion of the several types of trim that are set up in flight for the determination of static stability is given below.

For a given trim tab setting, e.g., engine power, and gross weight, there are potentially three conditions of trim for every c.g. location.

(*1*) *Free Elevator Stable Trim.* When the controls of a statically stable airplane are freed, the airplane's speed will tend toward its free elevator stable trim speed. If the airplane is dynamically, as well as statically, stable, it will eventually find its trim position at the stable trim speed. If the airplane is dynamically unstable but statically stable, it will oscillate about its trim speed with divergence. Since most airplanes are dynamically stable, the latter condition does not usually occur. The free elevator stable trim point is readily found by the pilot if the elevator friction is not too high. When the friction is excessive, some practice is required to find the elevator position corresponding to zero hinge moment.

(*2*) *Free Elevator Unstable Trim.* Most conventional airplanes with high powers lose stability as the air speed decreases. When the speed is decreased from the stable trim point, elevator forces will continue to increase to a certain point and then decrease. A point will occur where the elevator force again becomes zero. This point is the free elevator unstable trim speed. The airplane will not itself "find" this point when the controls are freed as in Part (1) above. If the airplane is set exactly on this speed and the controls are freed, the airplane, being unstable, will either increase speed to the stable trim speed or decrease speed until the stall occurs. Thus, in order to find the unstable trim point, the pilot will set speeds in the unstable region until he locates the point at which the speed, when slightly increased, will continue to increase and, when decreased slightly, will continue to the stall. High control friction again requires some skill on the part of the pilot. For some tab settings, unstable trim will not occur above stalling speed. This is discussed further in later sections of this paper.

(*3*) *Fixed Elevator Trim.* If the elevator control is held at a constant position, the airplane will assume a definite trim ($C_M = 0$) speed depending upon the airplane's c.g. location. Use of this trim condition is made for finding the fixed elevator stability, which is of value in comparing flight with wind-tunnel results.

"Effective Weight Moment" Flight Test Procedure

All of the trim procedures discussed above are used for the determination of the static longitudinal stability. The first two are used to determine the free elevator stability and the third for the determination of fixed elevator stability. If all three trim points are obtained for a series of c.g. positions, fixed and free elevator moment curves may be plotted.

Typical flight stability data are shown on Fig. 16. These curves are plotted in flight in order to facilitate the test and insure that the data being obtained are consistent. The significance of the curves is discussed

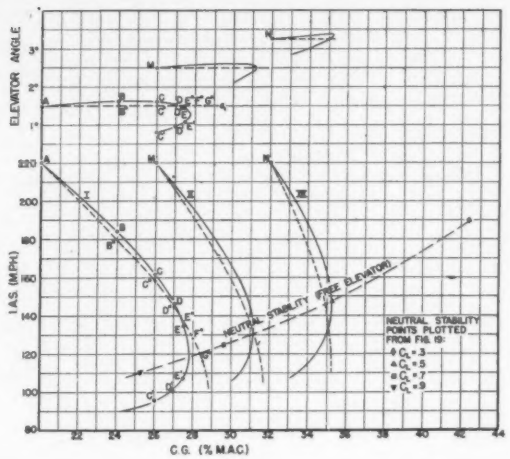

FIG. 16. Variation of trim speed and elevator angle with center of gravity location.

in a later section. The following discussion of Fig. 16 outlines the flight procedure used.

(1) The airplane is initially trimmed at a specified trim speed and specified c.g. position by means of the elevator trim tab:

(a). The initial trim speed, 220 m.p.h. in this case, is usually specified at some point in the high-speed range.

(b) The initial c.g. position, 20 per cent, is taken at the forward extremity of the range of loading conditions under consideration.

(c) Once the airplane is trimmed (point A), the elevator tab is locked and not moved until the complete curve AC′ has been obtained.

(2) The control column is then pulled back in the attempt to discover the unstable trim speed. For the typical case plotted Curve I, it is seen that at this c.g. the unstable trim point would occur below stalling speed.

(3) The c.g. is moved to 24 per cent and the free elevator stable trim speed is found (point B).

(4) The elevator is returned to its initial position, e_1, and the fixed elevator trim speed is found (point B″).

(5) The unstable trim point for 24 per cent is sought. As in Part (2), unstable trim does not occur above the stall.

(6) The c.g. is moved to 26 per cent and the three trim points, C, C′, and C″ are obtained as before. In this case unstable trim occurs above stalling speed (point C′).

(7) The c.g. is moved progressively aft and the points as shown on the curve obtained as before.

Power is held constant and altitude is restricted to as small a range as practicable during the determination of the curve. The trim points may be for diving, level, or climbing flight. For example, level flight might possibly occur at 162 m.p.h. for the power condition being tested. For this case, at points A, B, B″, the airplane is diving; at C, the airplane flies level; and for all other points, the airplane is in a climb.

Curves II and III are obtained by the same procedure. The only difference between Curves I, II, and III is that the initial trim c.g.'s (points A, M, and N) are different for each. Consequently, each curve is for a different (but constant) tab setting.

Fig. 17 is plotted in flight along with Fig. 16. $C_{M_{ac}}$ is a function of the c.g. position and may be obtained graphically. C_L is computed from the formula

$$C_L = W/{}^1/_2\rho_0 V_i^2 S$$

This calculation is easily made in the cockpit so that plotting of the $C_{M_{ac}}$ vs. C_L curve in flight is straightforward.

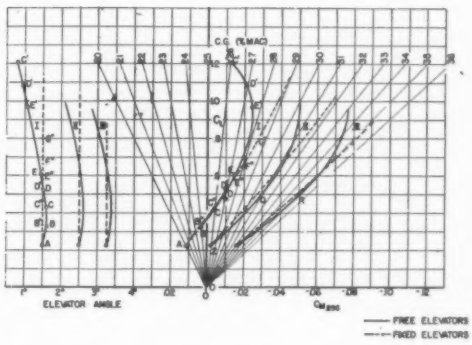

FIG. 17. Pitching moments and elevator angle versus lift coefficient.

Flight Test Problems

It is shown later that static longitudinal stability is a function of gross weight, altitude, engine power and efficiency, as well as air speed and c.g. location. The gross weight of the airplane is, of course, decreasing during flight as fuel is consumed. Therefore, as it is desirable to test at constant gross weight, the length of stability flights should be limited by the rate of fuel consumption. For any particular stability test program, an average gross weight is chosen; and an attempt is made to keep the gross weight in flight within 5 per cent of the average value.

It has been pointed out that data are taken during climb, level, and dive conditions. It is, therefore, impractical to set the altitude limits for tests too precisely. The normal range of altitudes within which the data are usually taken is between 4,000 and 10,000 ft., the lower limit being dependent upon the roughness of the air. Corrections are applied to the data to modify the results to constant "average" values of gross weight and altitude. The above-discussed limits on gross weight and altitude are considered necessary because the corrections are only approximate and tend to reduce the validity of the flight test results if large.

Shifting the c.g. One of the major functional prob-

lems of the type of stability tests described is to get an efficient system for moving the center of gravity in flight. Two methods have been used in the past: movement of lead ballast by a small crew, and movement of a large number of crew members themselves. The latter has proved to be the most satisfactory for an airplane of the heavy bomber type. When extremely wide c.g. limits are necessary for the tests, the two methods are combined. For small airplanes, this manner of moving the c.g. is obviously impossible; for large airplanes, it becomes more and more impractical as the gross weight increases. Water exchange systems have been used, but unless the pump is of an extremely high capacity, the time required to move c.g. by this method is prohibitive. For airplanes between about 25,000 and 80,000 lbs., the movement of crew members (each carrying as much ballast as possible) is satisfactory.

Trim Tab Problems. Usually in stability testing, it will not be possible to discontinue a curve and then to continue it later if, in the meantime, the trim tab has been moved. The accuracy of the tab indicators that have been used is of the order of $\pm 0.1°$. With the tendency of present-day aircraft toward lower control surface hinge moments, the trim tab effectiveness tends to increase; thus, slight elevator trim tab motion (presumably less than $\pm 0.1°$) will change the elevator position enough so that the trim speed is considerably different. For this reason the flight test procedure has been to complete the stability curves before any change in trim tab is allowed. Trim tab creep is, for the same reason, extremely critical. High friction arrangements on the tab control are used in the test airplanes to eliminate any possibility of creep.

A finite amount of lost motion which allows appreciable trim tab movement is usually present in tab hinges. It is easily possible that the aerodynamic loads on the tab will change direction during the course of a stability test (for example, at points C and C′, Fig. 16); therefore, rather than let the tab move through its lost motion angle, a bungee arrangement is installed on this tab. The bungee consists of a spring that holds the tab in one direction (normally the direction of the prevalent aerodynamic loads).

Miscellaneous Problems. Autosyn indicators are used for the indication of control surface positions. The transmitters are located at the surface so that cable stretch does not affect the reading. Gear trains are used to obtain sufficient accuracy, which should be 0.05°. Indicator dials are located both in the cockpit and in the photorecorder. The usual procedure is to install the instrumentation necessary for the tests both in the cockpit and at the photorecorder.

Sensitive type air-speed meters are used. Engine instruments, altimeter, and air temperature indicator are always included along with the instrumentation previously mentioned for stability testing. Data are recorded both by an observer in the cockpit and photographically. It is more practical to obtain the trim speeds in the cockpit, as some judgment may have to be used at the time. However, the photorecorder is used for elevator angle, power, altitude, and other data.

Test Data Reduction

The usefulness of the data gathered on an airplane stability test depends ultimately on its being expressed in some articulate form. The degree of static longitudinal stability of an airplane is measured by $(-dC_{Mc.g.}/dC_L)$. Numerous attempts have been made to set up a minimum limit of this factor for satisfactory stability of aircraft, although there are complications in the establishment of such a criterion: each type of aircraft requires a different amount of stability (for example, a fighter airplane probably needs less static longitudinal stability than does a long-range bomber); pilots do not consistently agree on what is "satisfactory stability"; a loss in controllability sometimes accompanies a gain in stability; and the balance between the two is a matter of pilot preference as well as demands of the particular airplane type. Rather than set up what is therefore an arbitrary value of $-dC_M/dC_L$, the value adopted as the comparison criterion is the point of infinite slope of the C_L vs. $C_{Mc.g.}$ curve (i.e., $dC_{Mc.g.}/dC_L = 0$, the point for neutral stability). This point corresponds to a definite airplane center of gravity; hence the final stability criterion is called the center of gravity for neutral stability (abbreviation, c.g.n.s.). This is of more actual significance in large airplanes where the loadings vary considerably.

Use of the neutral stability c.g. has several advantages in practice. The main stability variable over which there is any control is the placement of the useful load. If curves of c.g.n.s. are given, the stability picture for any loading is defined. Neutral stability is indicated also by the indicated air speed vs. c.g. curve; for neutral stability dc.g. $/d$i.a.s. $= 0$ (see Fig. 16).

It is desirable to obtain the c.g.n.s. for the entire flight and power range of an airplane. Considerable interpolation of the test data is therefore necessary because the actual measurements both in flight and in the wind tunnel are limited by time, expense, and other test difficulties. The i.a.s. vs. c.g. curve is not particularly suited for such interpolation because it is desired to keep C_L (or air speed) an independent variable. All interpolations of the test data are made from the C_M vs. C_L curves.

From the $C_{M.ac}$ vs. C_L curves, the c.g.n.s. is found. Let us assume that the three curves shown on Fig. 16 have been determined in flight for a constant power condition (e.g., rated power). The curves of Fig. 16 are first converted to $C_{M.ac}$ vs. C_L curves (Fig. 17). It is now desired to find the neutral stability c.g. for some specified C_L.

C.G. for Neutral Stability

For neutral stability, $dC_{Mc.g.}/dC_L = 0$. Therefore, if $dC_{Mc.g.}/dC_L$ be plotted vs. c.g., the point at which the curve crosses the $dC_{Mc.g.}/dC_L = 0$ axis will be the c.g. for neutral stability. The $dC_{Mc.g.}/dC_L$ vs. c.g. curves are plotted on Fig. 18 from the data of Fig. 17. Any point on the $C_{M_{ac}}$ vs. C_L curves may be converted into c.g. and $dC_{Mc.g.}/dC_L$.

$dC_{Mc.g.}/dC_L$ is determined by measuring the slope of the $C_{M_{ac}}$ vs. C_L curve and converting the value obtained by the usual equation given below.

$$\frac{dC_{Mc.g.}}{dC_L} = \frac{dC_{M_{ac}}}{dC_L} + (\text{c.g.} - 0.25)$$

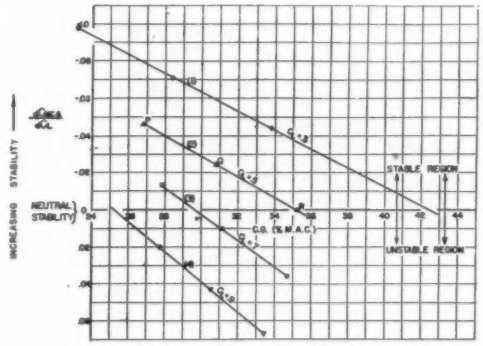

FIG. 18. Extrapolation for determination of the c.g. for neutral stability.

Normal procedure for test data reduction is to plot $dC_{Mc.g.}/dC_L$ vs. c.g. for several constant C_L's. Points P, Q, and R (for $C_L = 0.5$) are indicated both on Figs. 17 and 18 so that the relationship between the two figures may be seen. P, Q, and R are joined by a straight line (Curve 2, Fig. 18) and the c.g. for neutral stability at $C_L = 0.5$ is thus obtained where the curve crosses the $dC_{Mc.g.}/dC_L = 0$ axis. The c.g.'s for neutral stability at other C_L's are similarly determined (Curves 1, 3, and 4, Fig. 18).

Center of gravity location is one of the predominant stability variables. For power-off flight, it has been found that the percentage loss in stability is proportional to rearward movement of the c.g. in percentage m.a.c. This condition would be shown by parallel $C_{M_{ac}}$ vs. C_L curves or a 45° slope of the $dC_{Mc.g.}/dC_L$ vs. c.g. curves. Referring to Figs. 17 and 18 on which are plotted power-on data, it is seen that the $dC_{Mc.g.}$ vs. C_L curves are not parallel and that the slopes of the $dC_{Mc.g.}/dC_L$ vs. c.g. curves are less than 45°. In other words, the loss in stability with aft movement of the c.g. is not so great with power-on as with power-off. For power-on conditions, it is therefore necessary that at least two, and preferably three, moment curves be obtained so that the slopes of the $dC_{Mc.g.}/dC_L$ vs. c.g. curves may be found and the c.g.'s for neutral stability obtained.

In flight, stability data are taken for several different power conditions, for example: (a) propellers windmilling; (b) 40 per cent rated power; (c) 75 per cent rated power; (d) 100 per cent rated power. The c.g. for neutral stability for each of these powers is determined at several C_L's by the procedure outlined above. The neutral stability c.g.'s thus obtained are plotted versus a "power index" to give the static longitudinal stability variation for the airplane's entire flight range.

Power Index

The power index used in the analysis of stability data combines the variables that affect power-on stability (i.e., horsepower output, altitude, and gross weight) into one factor. The index is derived from a basic nondimensional thrust factor such that:

$$\Delta C_{TD} = T/qS \tag{1}$$

In Eq. (1) and the following equations, the notations used are:

T = total thrust
q = dynamic pressure = $\frac{1}{2}\rho V^2$ (2)
S = wing area
η = propeller efficiency
P = total engine output horsepower
ρ = density of air
ρ_0 = density of standard air = 0.002378 slugs per cu.ft.
σ = relative density = ρ/ρ_0 (3)
T = $(\eta P/V) \times 550$ (4)
C_L = W/qS or $S = (W/q)C_L$ (5)

Substituting Eqs. (2), (3), (4), and (5) into Eq. (1),

$$\Delta C_{TD} = \frac{\eta P\sqrt{\sigma}}{W^{5/3}} \times 550\frac{\sqrt{\rho_0 S}}{2} \times C_L^{3/2}$$

Finally:

$$\frac{\Delta C_{TD}}{C_L^{3/2}} = \left[\frac{\eta P\sqrt{\sigma}}{(W/1{,}000)^{5/3}}\right]\left[\frac{\sqrt{S}}{1{,}665}\right] = \text{p.i.}$$

This factor is called the "power index" and is similar to P_{tw} extensively used in performance.

$$\text{p.i.} = \left(\frac{1{,}000}{W_{std.}}\right)^{5/3}\frac{\sqrt{S}}{1{,}665}\eta P_{tw}$$

Since the variation of η is small compared to the range of P, T, and W, the index can be approximately computed for any flight condition. The power index reduces the flight conditions to a fundamental nondimensional variable representing the power effect on stability. Examination of the power index shows that the power effect is greatest at minimum gross weight, minimum altitude and maximum engine power.

Fig. 19 is a typical c.g.n.s. vs. p.i. plot for an airplane of conventional arrangement where the power effect is large and destabilizing. One plot of this type is made for fixed elevators and another for free elevators;

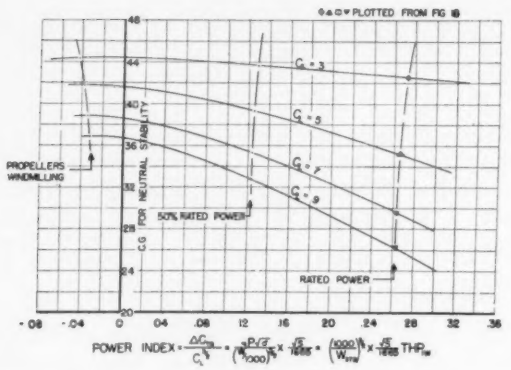

FIG. 19. Static longitudinal stability summary plot—free elevators.

thus, the complete stability picture of the airplane is given. From this curve, the minimum stable speed for any c.g. position may be determined. Conversely, the rearmost limit of the c.g. for stability at any designated speed may be found.

Figs. 16, 17, 18, and 19 plot idealized data and are included only to indicate the methods of flight testing and data reduction. Actual test data are given in subsequent figures.

Wind-Tunnel Tests

The study of stability characteristics can now be accomplished quite effectively in the wind tunnel prior to flight. In this manner the effects of various modifications can be examined exhaustively under controlled conditions and later confirmed by less extensive flight tests. Powered wind-tunnel models with self-contained motors and propellers are used for the measurement of longitudinal stability in the wind tunnel.[6] There are two major schools of thought as to the best technique to be used in the running of power-on wind-tunnel tests.

(1) Matching Full-Scale Conditions Directly: Specified full scale engine operating conditions are chosen before the wind-tunnel tests are begun, and all pitching moment data are taken by matching those conditions. This procedure necessitates matching the thrust for the condition against the C_L (or air speed).

The most critical conditions from the stability standpoint are usually chosen (for example, take-off power at a specified gross weight or rated power at a definite gross weight and altitude).

(2) Constant r.p.m. Polars: This technique consists of measuring pitching moments, lift, and drag for several different constant r.p.m.'s of the model motors through the propeller r.p.m. range from windmilling to maximum thrust r.p.m. The wind-tunnel curves obtained in this way can be interpolated to give any full-scale power condition required rather than the two or three definite conditions run by method (1).

The second method of wind-tunnel test for stability is preferred because it is felt that the complete data obtained in this manner are essential in order that the basic power effects may be thoroughly analyzed. The advantage of having complete data (i.e., all power conditions) is also of very great value in relation to flight test. It is desirable to determine the validity of wind-tunnel tests by comparable tests in flight. It is not always possible to test at the same conditions that had seemed likely when the wind-tunnel tests were run. If complete wind-tunnel data (i.e., method (2) above) were taken, the average flight condition may be matched by the wind-tunnel curves. In addition, the wind-tunnel data (which is much more complete than the flight data) may be used to greater advantage in reducing the flight results to a constant gross weight and altitude.

Comparison of Wind-Tunnel and Flight Stability

A considerable amount of comparable wind-tunnel and flight-test data has been obtained recently for both fixed and free elevators, and it has been found that, although the wind-tunnel tests of free elevator stability are completely inconsistent with flight tests, the agreement of the fixed elevator stability is excellent.

The lack of agreement of the free elevator stability as measured in the wind tunnel with the actual stability as shown by flight-test data is not surprising. The free elevator stability is a function of the trail angle of the elevator which is in turn dependent upon the direction of airflow at the tail. The induced effects of the wind-tunnel wall change the direction of the velocity vector, causing the elevators to trail at an angle different from that taken by the airplane elevators in flight. It is also a function of the thickness of the boundary layer which is different in the wind tunnel than in flight. The normal corrections applied to the moments to correct for the change in down-wash due to the tunnel walls do not take into account this change in elevator angle. The wind-tunnel wall correction is therefore applicable to the case of fixed elevators only. A suitable correction for free elevator wind-tunnel tests is perhaps possible by careful and complete analysis of hinge moment data. For the present, at least, the measurement of pitching moments with free elevator has been discontinued because of its irrelevance to the actual flight stability.

Figs. 20a–20f compare the moment curves obtained in the wind tunnel with those measured in flight. It has been noted that stability ($-dC_M/dC_L$) changes with trim; consequently it is necessary to compare flight and wind-tunnel curves at the same trim. In general, the wind-tunnel curves are taken at greater trim increments than are the flight curves, and therefore it is necessary to interpolate the wind-tunnel curves to the same region on the plot. A linear change in slope is taken. (This linear variation corresponds to a straight line fairing of the $dC_{Mc.g.}/dC_L$ vs. c.g. curves, Fig. 18.)

Figs. 20a and 20b show power-off and cruising power

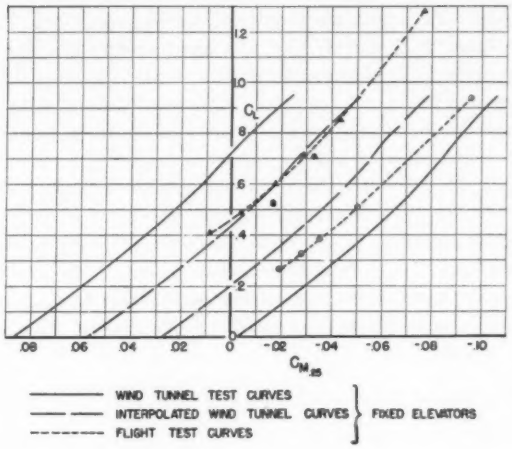

FIG. 20a. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane I, propellers windmilling.

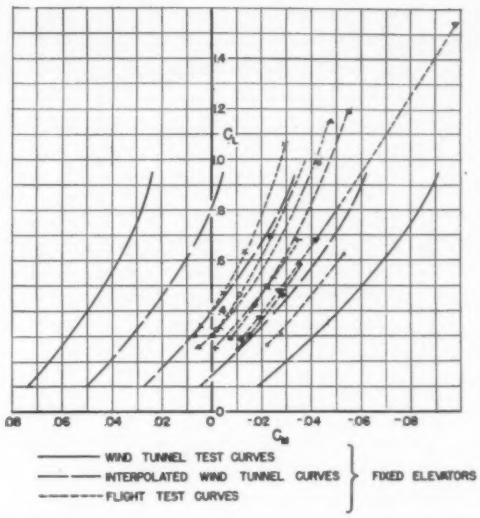

FIG. 20b. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane I, 75 per cent rated power.

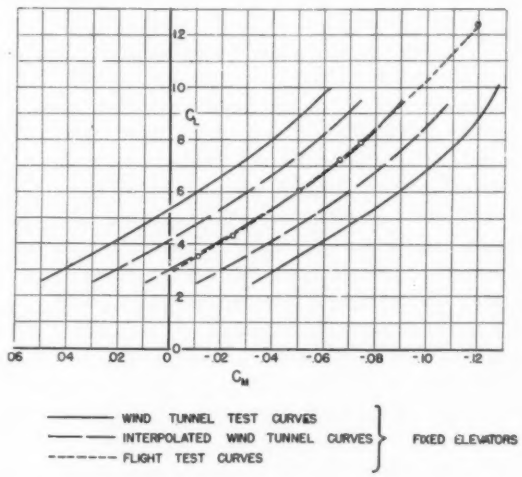

FIG. 20c. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane II, propellers windmilling.

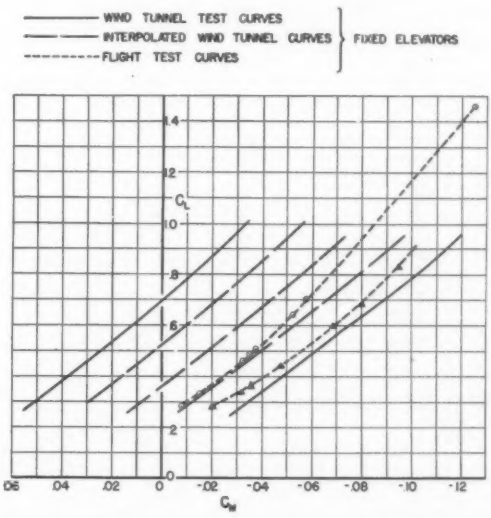

FIG. 20d. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane II, 40 per cent rated power.

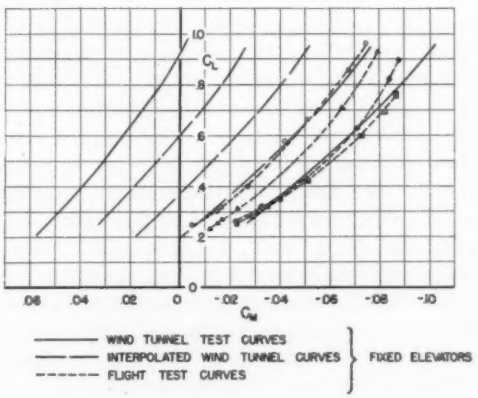

FIG. 20e. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane II, 75 per cent rated power.

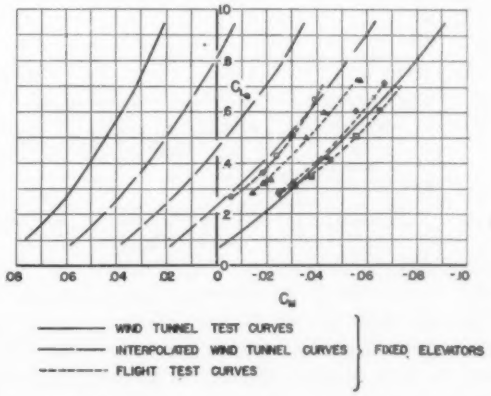

FIG. 20f. Comparable flight-test and wind-tunnel moment data—fixed elevators—Airplane II, 100 per cent rated power.

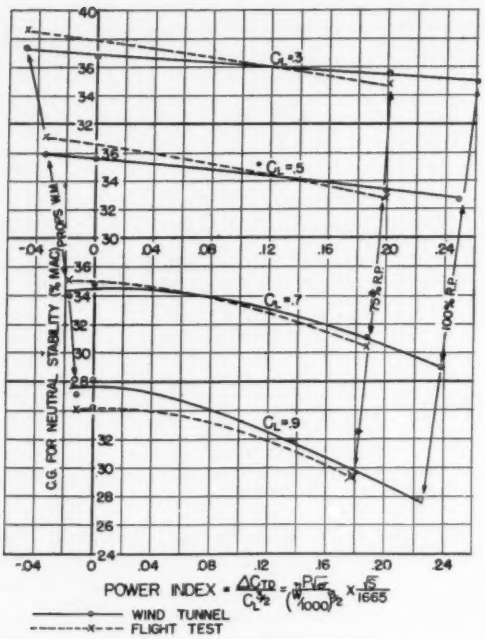

FIG. 21a. Comparable flight-test and wind-tunnel stability summary curves—fixed elevators—Airplane I.

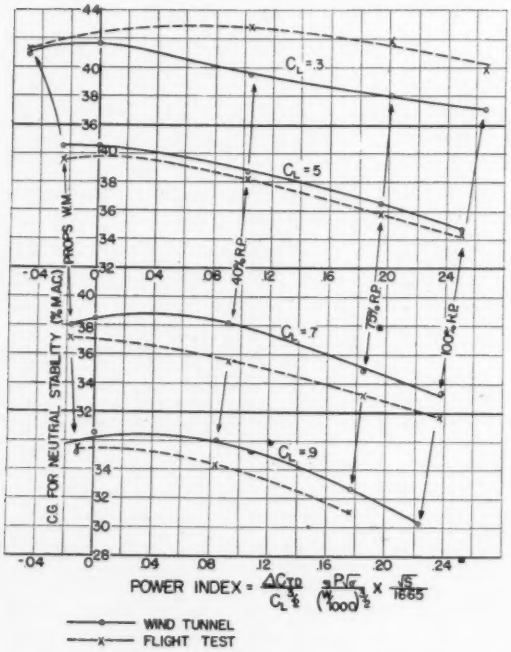

FIG. 21b. Comparable flight-test and wind-tunnel stability summary curves—fixed elevators—Airplane II.

stability comparisons, respectively, for Airplane I. Figs. 20c–20f show power-off, low, medium, and high powers, respectively, for Airplane II.

From the data of these figures, the center of gravity for neutral stability and the power index have been computed by the procedure discussed earlier in this paper. Plots of c.g.n.s. vs. p.i. are shown in Fig. 21a for Airplane I and in Fig. 21b for Airplane II. Both the flight and wind-tunnel data appear to agree within 2 per cent m.a.c. in general. The wind-tunnel data of Airplane II differ a considerably greater amount from the flight data at low C_L's than at high C_L's. This is the only case of definite disagreement shown. The discrepancy is due partially to the fact that the airplane cannot easily be flown at the extreme aft c.g. position necessary to get neutral stability at low C_L's; consequently considerable extrapolation of the $dC_{M_{c.g.}}/dC_L$ vs. c.g. curve is necessary at low C_L's (e.g., Fig. 18, $C_L = 0.3$).

Other Stability Tests

The measurement of static longitudinal stability is but one phase of stability-controllability testing. A complete treatment of all phases of the subject is the job for a text and, consequently, is beyond the scope of this paper. Mention should be made, however, of one or two other stability items and recent advances in flight test techniques for their measurement.

One tremendous step that has completely changed the concept of flight testing in the last decade is the development of accurate instrumentation and of automatic methods of recording flight-test data. Illustrations of the value of automatic recording in performance testing have been shown. Its use in stability tests is possibly even more pronounced. For dynamic conditions the adaptability of continuous automatic recording of pertinent items is immediately apparent. For example, the measurement of dynamic longitudinal stability is accomplished by setting up a longitudinal oscillation of the airplane and obtaining time histories of air speed, altitude, elevator position, trim angle, etc., from the automatic recorder results.

The adaptability of automatic recording to cases where static or trimmed conditions are tested is not immediately evident. The use to which it may be put for tests of this type is best illustrated by a typical example: the rudder force and lateral stability test.

For this test it is desired to obtain curves of rudder force versus rudder and yaw angles for conditions of symmetric and unsymmetric power (i.e., different yawing moments). The standard procedure for such tests in the past has been to set a specified rudder angle and then, by means of the ailerons, to balance out the roll due to rudder deflection before taking data. Thus, one test point would be obtained for each rudder angle set.

A rudder test technique that has been developed by the use of automatic recording makes one assumption: that static conditions are simulated by extremely slow maneuvers. Rather than trim the airplane at each of the desired rudder deflections, the rudder is moved slowly from neutral position to maximum position and at the same time the wings are kept level

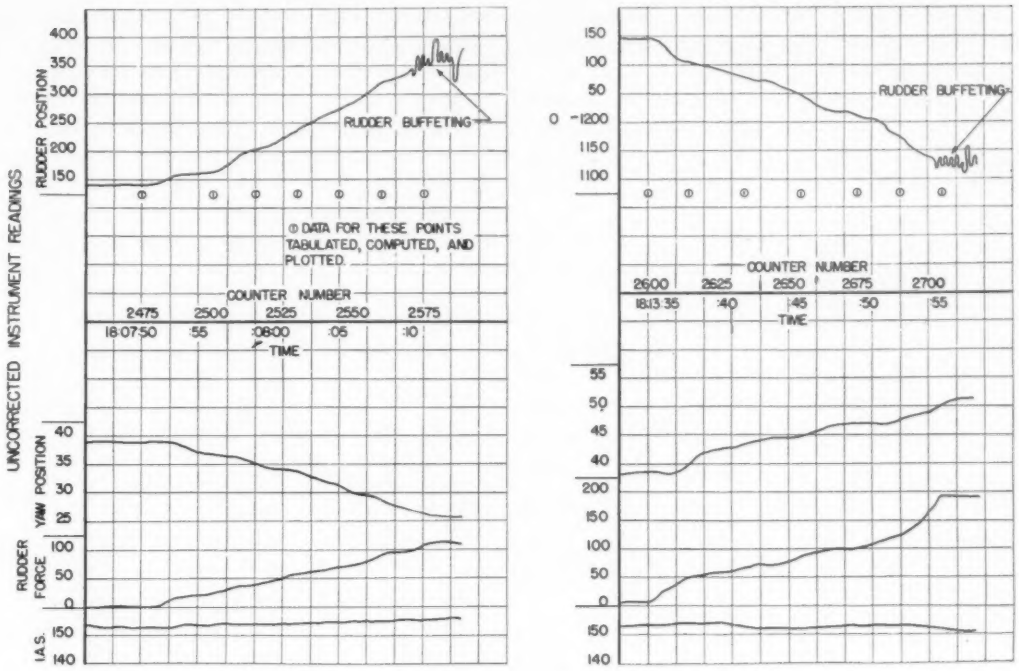

FIG. 22. Rudder flight-test data—time history of variables.

by the use of ailerons. The time taken to complete the maneuver (that is, to reach the rudder stops or the maximum pilot force) is approximately 20 sec. Data are taken continuously by the automatic recorder during the maneuver so that a complete time history of the variables is obtained. Fig. 22 plots typical data obtained in this manner. The time histories of rudder position, rudder force, airplane yaw, and indicated air speed are shown in the figure. Other items that may be taken at the same time are:

(1) *Angle of bank, rates of roll and yaw.* These variables should be measured to check the finesse with which the maneuver was performed.

(2) *Aileron angle and force.* Aileron forces and positions for such a maneuver are valuable data.

(3) *Direction of flight (directional gyro).* It is of interest to find the path of the airplane at zero angle of bank and zero velocity in roll.

The data of Fig. 22 were recorded at approximately $^1/_4$-sec. intervals, and hence all disturbances (even those of very short period) are clearly shown by the plot. Rough air is immediately apparent (the air for the test shown was relatively smooth). Rudder buffeting tendencies are shown graphically.

The rudder test data obtained by the new method show considerable improvement over those obtained by the old method, both qualitatively and quantitatively. Fig. 23 is a typical plot of the reduced data, showing rudder hinge moments and airplane yaw versus rudder angle. The degree of scatter is much less than that obtained by the old method of testing.

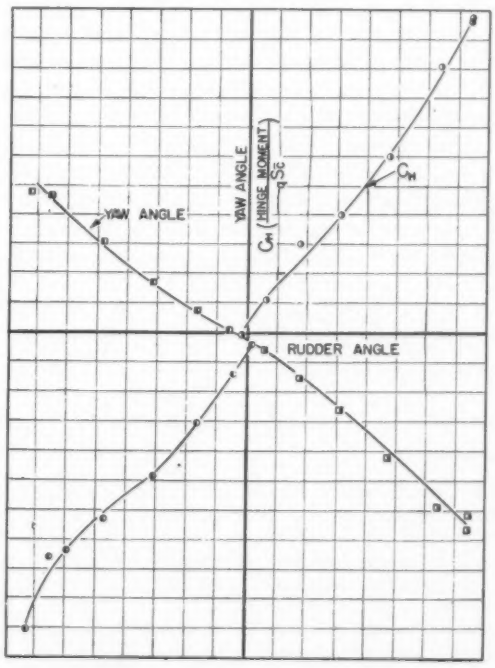

FIG. 23. Rudder hinge moment and airplane yaw angle vs. rudder angle.

The number of test points used in plotting the curves is arbitrary; twice or three times as many as shown are available if scatter is bad. The manner of reducing the data is as follows:

(1) The uncorrected, uncalibrated (photographically recorded) data are plotted versus time (and/or) counter number as shown on Fig. 22.

(2) Points are chosen at random along the abscissa scale (see Fig. 22) and the uncorrected instrument readings (rudder angle, yaw angle, rudder force, i.a.s.) corresponding to those points are tabulated on computation sheets.

(3) Calibrations and corrections are applied reducing the variables to the form in which they are to be plotted.

If the plots of these points indicate that more data are desirable, the same procedure may be applied to other points. The typical flight-test data obtained by this procedure (Fig. 23) are considered excellent; maximum deviation of test points from the curves faired-in is approximately 3 per cent of the total magnitude of the variables.

These examples are typical of the procedures used in many other phases of present-day flight testing. They are indicative of the rigorous and careful approach that is being used throughout.

Conclusion

Flight testing of aircraft is becoming an exact science rather than the almost entirely qualitative study it was only a few years ago. No longer is the pilot's personal reaction to the "feel" of an airplane the sole basis upon which, for example, the aerodynamicist must modify a control surface. Pilot reaction to an airplane's flying qualities is still of first importance. When integrated with accurate quantitative data, however, the value of the pilot's qualitative report is tremendously increased.

We now have an accurate method of analysis of flight-test data based on the physical laws of fluid mechanics. This method of analysis has been so illuminating that it has opened up a whole new field of flight-test technique and instrumentation. As a result, we now measure many of the formerly uncontrolled variables of a test flight and manipulate these measurements into a closely knit mathematical expression of the airplane's capabilities. Aeronautical developments of a tremendously accelerated research program are now awaiting flight-test verification. Changes in design dictated by war and commercial flying experience must be tested. In many cases the knowledge gained from a series of flight tests enables us to predict flight characteristics that call for a new series of exploratory or confirming tests.

The need is present, the tools for doing the job are at hand, the techniques for using the tools have now been developed both for performance and stability testing. Certainly during the next few years, the current advances in the field of flight testing will be felt more and more as a highly influential factor in our future aeronautical progress. The work that Orville and Wilbur Wright began with their first test flight 39 years ago is being carried to a new level in the spirit in which they began it—the spirit of the scientist seeking by a painstaking step-by-step process of experimentation and analysis to fill out gradually our field of knowledge which will permit still further design improvement.

Acknowledgment

I wish to give grateful acknowledgement for the assistance in the preparation of this paper of William Cook, Vincent North and Lloyd Goodmanson, as well as to a number of other aerodynamicists and pilots who have been kind enough to read and contribute to the manuscript.

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Discussion of the Lecture

Mr. Charles H. Colvin, Director of the Daniel Guggenheim School of Aeronautics, New York University, acted as Chairman of the Lecture. At the conclusion of the Lecture Mr. Colvin called for the reading of the following prepared comment on the subject:

Dr. Norton B. Moore, Airplane Division, Curtiss-Wright Corporation: "Mr. Allen is to be congratulated on his excellent exposition of the 'aerodynamic approach' to flight testing. The results speak for themselves in attesting to the wiseness of Boeing's consolidation some years ago of Aerodynamics and Flight Test under Mr. Allen's able direction, separate from engineering, sales and other departments.

"It is difficult to 'discuss' such a Lecture when one is enthusiastically in agreement with practically everything in it, but my colleagues and I do have a few comments to offer here:

"The value of parallel testing in flight and wind tunnel, emphasized in the Lecture, is rapidly becoming more widely realized. Company wind-tunnel projects, the magnitude of which not so many years ago might have been considered fantastic, when in use in the near future in conjunction with scientific flight testing will undoubtedly play a large part in improving the quality of United Nations' planes.

"We would like to add a note of caution concerning the assumption that all airplane polars can be expressed in the form $C_D = A + BC_L^2$ over a reasonably wide range of lift coefficient. Wind-tunnel tests should be used as guides in this matter, inasmuch as some flying boats deviate markedly from a parabolic polar and, even in the more usual landplane type, 'buckets' may occur in the lift-drag curve, due to peculiar wing section characteristics, local stalling of engine cowls, etc.

"I am a little surprised at the use of an empirical method of correction of propeller data for the purpose of bringing lift-drag points on to a parabolic polar; however, until more propeller data are available this is no doubt justified.

"Mach number effects are stated to be negligible on *most* present-day airplanes, but I think it will not be very long before we will all be much more closely acquainted with compressibility effects on performance. Thrustmeter development and further experience with compressibility effects, both on airplanes and on propeller, should lead to still better correlation of data and to design advances.

"It is to be hoped that the torquemeter will be recognized by the operator as being just as useful to him as it is to the flight-test pilot and engineer.

"The importance of careful calibration, and frequent recalibration, of instruments cannot be overemphasized, particularly to novices in flight test work.

"Meticulous weight control and weight correction in data reduction are still too often neglected in many cases for small or high-speed airplanes. Weight is a prime performance factor whether the plane be large or small, and its effect is a function of the basic performance parameter,

$$\Lambda_\sigma = W^2 f^{3/5} / eb^2 THP^{4/5} \sigma^{3/5}$$

(*cf.* N.A.C.A. Report No. 654). Thus flight-test data taken on high-altitude rate of climb of a pursuit or on single-engined rate of climb of a small twin-engine trainer may be worthless unless the weight-time relationship is measured and taken into account.

"The '*iw*' method of data reduction will, we hope, become standard practice in all companies and government agencies, even though it as yet cannot handle Mach number effects and even though we find N_{iw} to be a somewhat cumbersome parameter for correlation with propulsive efficiency.

"It is to be regretted that time allowed only one sentence on partial engine performance. Partial engine performance flight testing techniques are usually such that it is difficult to get good results, and I wish some discussion of good methods for, and cautions regarding, this kind of testing could have been included.

"Take-off and landing tests might well have been mentioned also had time permitted. There is no comparison between the old approximate measurements and the results obtainable with modern equipment, such as the Bell and Howell flight analysis equipment which we use, with which we can determine just where the airplane has changed attitude, has hit a bump on the runway, etc., and whether braking has been steady or erratic, for instance.

"Mr. Allen's three rules concerning weather might well be framed and hung in every flight-test hangar.

"The discrepancies in rates of climb shown where lapse rates are not smooth are most interesting and proably merit further investigation.

"I wish some time could have been given in the Lecture to elevator force and position measurements for stability, for although these are straightforward, as Mr. Allen states, they are by no means easy. And they are most important for pursuit and similar airplanes, where maneuverability and controllability are essential. Although the statement is made that flying qualities can be established quantitatively as accurately as can performance data, this does not appear to us to be completely true. Most attention is given in the Lecture to determination of pitching moment curves, but this is only one part of flying characteristics.

Such things as stick loads and damping under high-load factor, aileron loads with high values of $pb/2V$, etc., are just as important as are steady state pitching moments for some classes of airplanes.

"In connection with the two schools of thought on powered model testing, we use method (1) since the way we make our tests, it has the advantages of (1) and (2). Usually at least four to six airplane power conditions are selected to furnish a reasonable variation in P_{tw}, and tests are made for these conditions. The results are then cross-plotted for other conditions. However, it has been found that cross-plots from constant model power polars may not be satisfactory for obtaining slopes of pitching moment curves unless a large number of constant powers is used. This is due largely to the fact that ΔC_{TD} is not linear with C_L, hence at large C_L's few points are available in the cross-plots.

"Introduction of the 'power index' is interesting, but it is our feeling at Curtiss-Wright that the THP_{tw}, which we have used for several years in stability as well as performance work, is more convenient than the dimensionless 'power index.' It is an 'effective' power, but it does not necessarily measure the *effect* of power on stability. The statement that '. . . the power effect is greatest at minimum gross weight, minimum altitude, and maximum power' may be misconstrued. Under these conditions the power index or the THP_{tw} is, of course, a maximum, but it does not follow that the effect of power on stability necessarily increases with increasing power. We now know how to design three types of airplanes in which the effect of power on stability is completely nil, or is stabilizing, or is destabilizing. And in the usual case where the power effect is destabilizing, such as in Fig. 19, in some airplanes a point is reached where further application of power results in no further change in stability.

"Lack of agreement of free elevator stability in wind tunnel and flight is, I believe, a challenge to the wind-tunnel people. We have hopes of better correlation and are working toward that end. Better elevator-free results are obtained if constructed from zero hinge moment points from elevator-fixed tests in the tunnel when hinge moment polars are run. Several factors concerning elevator free runs in addition to those mentioned in the Lecture enter into the problem, an important one being the effect of longitudinal pressure gradient on model elevator-free trail angle.

"The Institute is to be congratulated on its choice of Mr. Allen as the Sixth Wright Brothers Lecturer."

W. K. Ebel, The Glenn L. Martin Company: "The flight testing of airplanes has certainly changed in many respects since Wilbur and Orville Wright made their test flights 39 years ago today. The longest flight of that day, 852 ft., was about half the take-off run of the heavier class of airplanes that are being produced today. The take-off time of a modern flying boat is about the endurance record for the day of the world's first successful flight test—59 sec.

"The Wright Brothers said little about how they carried on their flight test program. The romance is gone from flight testing and it has become a tough grueling routine. The tests are planned, executed and reported by men especially trained for their own particular phase of the work. In these days of increased tempo, the Institute appreciates the time Mr. Allen has taken from his work to write about how he does it. The industry has been cognizant of the thorough manner in which Mr. Allen has conducted flight testing and will benefit by learning more about his procedure. The complement to his excellent exposition of 'Flight Testing for Performance and Stability' will be a paper soon to be read before the Institute by Mr. W. B. Bergen on 'Experimental Investigation in Aircraft Dynamics,' dealing with flight testing for structural integrity.

"A certain portion of the flight testing done by the aircraft manufacturer is done in order to be able to instruct the user upon the manner in which the airplane is to be operated. Mr. Allen brings this out when he states that airplane performance can be improved not only by design but by controlling the method of operation. The difference between *maximum* performance under test conditions and *probable* performance under operating conditions is far too great, particularly in view of the abbreviated training now available to the men who are operating the planes. Much work remains to be done to either simplify efficient operation or to provide better training in efficient operation.

"The engineers of the Martin Company have been able to obtain fair correlation between wind-tunnel and flight-test data for control forces for elevators and rudders and expect to get comparable correlation for aileron forces. This is done by testing the complete model with fixed control surfaces at several angles and then measuring hinge moments on larger size models of the control surfaces and the fixed surfaces to which they are attached. By combining the information from these two series of tests, it has been possible to obtain the desired control force characteristics with only minor changes in control surface contour.

"Mr. Allen's charts show the effect of Mach number on the performance of the propellers but do not show its effect on the drag of the airplane. This is a question much discussed among aerodynamicists, and opinions are widely divergent. There is now a lack of reliable flight-test data on the subject of variation of drag coefficient with Mach number.

"Mr. Allen has written an excellent paper in which he has clearly told how the major problems of flight testing are solved and is to be heartily congratulated for it. This discusser entertains the hope that he may expound upon the subject further and make the information available to the industry."

Paul S. Baker, Vought-Sikorsky Aircraft Division, United Aircraft Corporation: "It is indeed a privilege to hear from such a complete authority on the subject of flight testing as Eddie Allen. I can think of no individual who at the same time can better speak of both the engineering and practical flying aspects of the subject from his own knowledge and experience.

"The paper treats of what might be termed the 'routine' phases of performance and stability testing—that is, the determination of the characteristics of a given airplane for quantitative expression. One is immediately struck by the amount of testing which must be accomplished for a real determination of the characteristics of a large weight-carrying airplane throughout the possible ranges of gross weight, center of gravity location, altitude, power, and so forth. However, as the importance of this type of testing has become more and more evident as a necessity for efficient use of the equipment, and it is really important because of the wide range of possible operation conditions, which in turn must increase the scope of the tests, this necessity has been realized, and the testing has become 'routine.'

"To those of us concerned primarily with smaller airplanes with more restricted weight and center of gravity ranges and more limited cruising requirements, whose chief concern has been with what may be termed development testing, routine testing of the magnitude described seems most impressive. When it is considered that the Boeing organization must also be conducting extensive development work, then the amount of flight testing handled by them must be viewed with great respect.

"With regard to performance testing, each testing organization has its preferred methods of analysis and reduction to standard. Any method of analysis and reduction to standard conditions is based on certain assumptions as to the important variables and as to the less important and consequently negligible variables. While a fundamentally sound method of analysis is of the utmost importance, the other requirements for accurate flight testing must also be fully appreciated and satisfied, as the Lecture has pointed out. Under these conditions, apparent lack of agreement with the expected or scattering of points can be further investigated and apparent mysteries solved. To my mind, it would be difficult to overemphasize the importance of controlling the weight and configuration of the airplane and the other factors which influence the quality of the test data used for subsequent analysis, and I am glad to see them given the importance they deserve.

"In regard to stability testing, the procedure used to arrive at the neat set of curves of c.g. location for neutral stability against power index for various lift coefficients appears to be most direct. However, for those of us whose interest may be in smaller airplanes, shifting the c.g. in flight is a rather difficult problem, even when given the choice of moving lead ballast with a small crew or moving a large number of crew members themselves, especially on a two- or three-place plane!

"I am happy to see the importance of instrumentation emphasized. Great progress has been made in this direction in the past few years. The time when only the large government testing establishments had the necessary instruments for properly obtaining the necessary data for quantitative evaluation of results is past.

"In this connection, it is interesting to note the increasing use of automatic recording equipment, designed to take down more data in flight than can be recorded by the pilot or observers. It has sometimes been said that such equipment (on smaller planes) is designed to 'relieve the pilot (or observer) of the burden of laboriously writing down the data.' Actually, the use of such equipment increases the responsibility of the pilot or observer, if anything, for the pilot must now be sure that he makes all the necessary notations so that the records can be properly interpreted. Otherwise, the automatically recorded data cannot be properly interpreted, for no such automatic equipment can record all the necessary information.

"In conclusion, I wish to congratulate Mr. Allen on his excellent paper and compliment him on his organization, the work of which is reflected in the paper."

Melvin N. Gough, National Advisory Committee for Aeronautics: "The topic 'Flight Testing for Performance and Stability' by E. T. Allen, chosen for the Wright Brothers Lecture this year, is timely. This presentation makes more understandable the excellent performance of large American aircraft in the present world conflict. It clearly adds to the existing literature undeniable proof that such performance has been obtained not by chance but by thorough, painstaking, untiring research on the part of many persons, among the foremost of whom is the author. The major role taken by Mr. Allen in the development of large aircraft in this country is known the world over. He writes with authority. His scientific approach and the organizational development inspired by him focus attention on the part that flight testing and flight-test personnel can and do play in aviation development throughout the country.

"It is hardly to be expected that all the experiences of the author or all phases of the subject could be covered in detail in the present paper. Similarly, it is obvious that few organizations, private or governmental, could afford the extensive preparations, the great attention to detail, and the time required for such work. Although the principles suggested are sound, the completeness with which the details are covered must necessarily be limited. The application of the principles vary as the size of the aircraft under discussion varies

from those with which the author has been more recently associated.

"The airplane polar method of level-flight data reduction, although not entirely new, and the reduction of many performance characteristics to basic coefficient form are both too little used and are worthy of much greater consideration by flight-test groups working toward the goal of standardization. The size and type of airplane and the compressibility range considered by the author must, however, be kept in mind; aircraft types now contemplated will require new assumptions, particularly in the case of the polar and the allocation of Mach number effects. Recent tests, for instance, indicate that there may be considerable doubt that serious compressibility effects are occurring on the propeller at tip Mach numbers of the order of 1.0. A good point, which probably cannot be overstressed, is the necessity and difficulty of close control over airplane weight and external configuration. It is interesting to note that, whereas most flight tests must be conducted with center-of-gravity changes made between flights, the test pilot of a transport or bomber has at his instant command the placing of human movable ballast.

"While undoubtedly well known to the author, it is surprising to find so little attention given to the determination of fuel–air ratio and mixture distribution which, unless known, render engine temperature and cooling-air pressure drops practically meaningless. The increased demand for the use of the torquemeter is noteworthy, however, although for satisfactory results details of the principle of operation must be thoroughly understood.

"The obtaining and reduction of climb data as indicated by the paper still seems to remain a most difficult task and little clarification is forthcoming.

"The absolute necessity for skilled flight-test crews and the results possible are demonstrated by the author and need no further mention.

"In connection with the discussion on air-speed measurements, it would be interesting to have the author's views with regard to placing the air-speed head in such a position as to reduce calibration and position errors, use of a fully swiveling static-head in combination with a total-head tube, calibration by means of a trailing bomb, etc., the details of which could not be reasonably included in the present paper.

"The attempt to evaluate stability in terms readily understandable to the designer is commendable In passing, it is difficult to realize that any pilot could think in terms of $-dD_M/dC_L$ because of the complication of the many other factors that affect the stick force and position variations experienced by him. Experience in this country and abroad indicates complete agreement with the author on the point of the desirability of determining the neutral point or center of gravity for neutral stability.

"It is not surprising that mention is made of the inconsistency of correlation between flight and wind-tunnel tests in the case of elevator-free stability. Great care must be taken to guard against obtaining too much useless data when conducting such extensive tests as are indicated. In the case of control surfaces, for instance, considerable disagreement can be obtained between model and full-scale tests which may not be generally appreciated because of small normally undetectable geometric differences such as rib stitching, trailing-edge thickness and shape.

"Under the heading of 'Other Stability Tests,' mention is made of the use and development of automatic-recording instruments for obtaining continuous time-synchronized records of maneuvers. Such instruments have been used for years by the N.A.C.A. and have made possible the well-known collection of handling-qualities data instituted by Dr. E. P. Warner and currently obtained by the N.A.C.A. on many types of aircraft. Thus, stability and control characteristics are taking shape in quantitative terms, and pilots' impressions are being reduced to a rational engineering basis. It is agreed that by this technique the concept of flight testing has undergone tremendous development.

"It is to be hoped that the developments and progress indicated in this paper, and similar work being conducted elsewhere, will produce a better general understanding of flight testing and that by proper simplification of the principles involved the methods will be practically applicable to the large number of smaller aircraft establishments so essential but less lavishly endowed. The author is to be commended for a most excellent and interesting paper."

A. Lewis MacClain, Pratt & Whitney Aircraft Division, United Aircraft Corporation: "Flight testing is here evidently presented from the point of view of the airplane, or should we say air frame. There are numerous other items included in the flight-test program, and particularly in the flight testing carried out by the engine, propeller, supercharger and other accessory manufacturers. Performance and stability of performance are of prime importance to these other items, and the results are as essential in the overall design of the airplane as are the characteristics of the air frame.

"The lecturer has brought to our attention many of the refinements in flight testing that are now being used and gives a picture of flight testing that never once brings to mind the popular conception of the harum-scarum test pilot who pulls wings off airplanes. We get a good picture of the modern flying laboratory with its crew of engineers who are out on serious work.

"I agree with Mr. Allen that the pilot and copilot must be competent and sympathetic toward the type of testing that is being done. In addition, it is highly desirable that the pilot be an engineer taking an active

engineering part in the ground work concerned with the test. The engineer pilot, because of his active 'feel' of the airplane during the test, can be of as much assistance in analyzing the data as is the engineer observer. The necessity of a fully trained and integrated engineering crew thoroughly conversant with the instruments and equipment of the airplane under test cannot be emphasized too highly.

"As to wind-tunnel tests and flight tests, it sometimes seems that wind-tunnel tests are so well controlled that many items attendant to actual operation are difficult to introduce properly into the wind-tunnel tests and predict what will happen in flight. It is difficult for the human mind to visualize all of the ramifications attendant upon the operation of an airplane and all of its parts and to integrate properly from ground tests all of the interrelated performances of the component parts in order to obtain the overall performance. Sometimes there are tendencies to get more out of wind-tunnel tests than they can possibly give; for instance, it is difficult to put together the wind-tunnel tests of the air frame, the characteristics of the propeller, and the engine and carburetor characteristics and come out with a satisfactory schedule of performance characteristics for range at various altitudes and load conditions. The question of extrapolation from wind-tunnel models, particularly of large airplanes, to full-scale values arises. The use of extremely small models is not felt to be reliable because of the relationship of boundary layer and model size. A great deal of judgment has to be used in interpreting data obtained from either wind-tunnel or flight tests and the crews that make wind-tunnel tests and those that make flight tests should get together on their common problems.

"In addition to the difficulties and differences of reducing airplane performance to N.A.C.A. or I.C.A.N. standard conditions by the Army, Navy, and British methods—Mr. Allen overlooked the C.A.A.—there are the cooling requirements for so-called summer standards different for each agency with different correction factors for many items. Also there are the tolerances allowed for production which will cause individual airplanes to exhibit different characteristics in flight.

"In connection with the problem of propeller performance, progress is being made in measuring propeller thrust by finding the pressure head in the wake with radial rakes. This method is expected to be useful with high-speed airplanes in showing what part of the propeller is running into compressibility trouble. This method can also be expanded to find the effects of blade shape, pitch angle distribution, and other items affecting propeller performance.

"The lecturer mentions that on most present-day airplanes any effect of Mach number on drag may be assumed negligible. Several of the present-day fighters are operating at speeds that surely must induce high local velocities over certain portions of the wing and fuselage structures, and the effect of Mach number upon drag must begin to receive consideration.

"The matter of measurement of free air temperature and the determination of the percentage of adiabatic temperature rise is of importance. It enters particularly into problems concerning heat transfer from the engine, intercoolers, oil coolers, to the cooling air, and the effect of the superchargers upon the engine air. Checks upon the free air temperature thermometer installation are made while making air-speed calibration runs. Trials are being made in flight tests of 'total temperature' heads that have been developed in conjunction with supercharger testing. Laboratory tests indicate that these heads will measure better than 99 per cent of the total temperature, that is, the static temperature plus the adiabatic temperature rise.

"The rules given by the lecturer to assure satisfactory weather conditions for performance tests are excellent but may not be wholly satisfactory. It is felt that there are large areas or masses of air with upward or downward components of velocity, or else where does the air come from that flows outward from the high-pressure areas, and where does the air go which flows into the low-pressure areas of the weather maps? The vertical air-mass movement is not shown on the weather map, and usually smooth air is found where downward movements of air might be expected and low airplane speeds are often encountered. The long-distance flights of gliders show that considerable energy is at work in the atmosphere. The effect of this energy upon the powered airplane is difficult to determine.

"The subject of airplane stability in its finer points is outside the normal field of study of the engine manufacturer and he gladly turns this work over to the airplane flight-test groups and trusts their results. The engine builder finds enough 'hot' problems in the powerplant installation to keep him busy. The lecturer has given an interesting and illuminating account of certain current trends in the testing and analysis of air frame stability problems."

Clark B. Millikan, California Institute of Technology: "It is a very great pleasure indeed to felicitate both the Institute for its choice of Mr. Allen as 1942 Wright Brothers Lecturer and the lecturer himself for his remarkable paper. It has been a matter of common knowledge among aeronautical engineers that Mr. Allen's group has, during the past several years, been doing extremely important and valuable work in the field of flight testing. I suspect that it was only because of the significance of the present occasion that the speaker felt able to take the time from his pressing duties to collect and prepare some of his results and conclusions for public presentation.

"The magnitude of the field treated by Mr. Allen is indicated by the number of questions which his paper

must raise in the mind of every trained engineer. It is clear that a series of volumes would be required if all were to be dealt with. Three that have particularly interested me are the following:

"(1) In obtaining data for the P_{tw} vs. V_{tw} curves such as those of Figs. 1 and 2, it would appear to be difficult to make observations at exactly constant values of N_{tw}. There must then be a considerable amount of cross-plotting involved in the preparation of such figures. The reliability and accuracy of the results obtained from such cross-plotting procedures depends greatly on the specific techniques employed, and it would be most instructive to know more details of the methods found most satisfactory.

"(2) The nature of the 'equivalent propeller charts' mentioned by the speaker, as well as the scatter of the experimental points on them, would also be of great interest.

"(3) It is obvious that Mr. Allen's methods make possible much greater precision in the matter of climb reduction than has been usual in the past, but here again one's appetite for details is whetted but hardly satisfied.

"It is naturally a matter of considerable gratification that Mr. Allen has so satisfactorily discussed a subject that I had the privilege of presenting at an earlier Wright Brothers Lecture, namely, the effect of running propellers on static longitudinal stability. Three years ago I emphasized the desirability of systematic studies of the correlation between wind-tunnel and free-flight observations of the phenomenon. Mr. Allen has given examples of such correlations, and the highly satisfactory nature of the agreement found in the elevator-fixed case should be a source of profound satisfaction both to designers and to those engaged in wind-tunnel testing. The lack of agreement in the case of free elevators may serve as a stimulus to wind-tunnel experimenters to refine or extend their technique. Certainly Mr. Allen's paper will stimulate every alert worker in the flight testing field."

The 1914 Tests of the Langley "Aerodrome"*†

C. G. ABBOT‡
Smithsonian Institution

(The following statement by the Smithsonian Institution was published to counteract some previous impressions regarding the flying of the Langley machine at Hammondsport in 1914. Inasmuch as this controversy has an importance in the history of American aeronautics, it is reprinted in full.)

NOTE—This paper has been submitted to Dr. Orville Wright, and under date of October 8, 1942, he states that the paper as now prepared will be acceptable to him if given adequate publication.

IT IS everywhere acknowledged that the Wright brothers were the first to make sustained flights in a heavier-than-air machine at Kitty Hawk, North Carolina, on December 17, 1903.

Mainly because of acts and statements of former officers of the Smithsonian Institution, arising from tests made with the reconditioned Langley plane of 1903 at Hammondsport, New York, in 1914, Dr. Orville Wright feels that the Institution adopted an unfair and injurious attitude. He therefore sent the original Wright Kitty Hawk plane to England in 1928. The nature of the acts and statements referred to are as follows:

In March 1914, Secretary Walcott contracted with Glenn H. Curtiss to attempt a flight with the Langley machine. This action seems ill considered and open to criticism. For in January 1914, the United States Court of Appeals, Second Circuit, had handed down a decision recognizing the Wrights as "pioneers in the practical art of flying with heavier-than-air machines" and pronouncing Glenn H. Curtiss an infringer of their patent. Hence, in view of probable further litigation, the Wrights stood to lose in fame and revenue and Curtiss stood to gain pecuniarily, should the experiments at Hammondsport indicate that Langley's plane was capable of sustained flight in 1903, previous to the successful flights made December 17, 1903, by the Wrights at Kitty Hawk, N.C.

The machine was shipped to Curtiss at Hammondsport, N.Y., in April. Dr. Zahm, the Recorder of the Langley Aerodynamical Laboratory and expert witness for Curtiss in the patent litigation, was at Hammondsport as official representative of the Smithsonian Institution during the time the machine was being reconstructed and tested. In the reconstruction the machine was changed from what it was in 1903 in a number of particulars as given in Dr. Wright's list of differences which appears later in this paper. On the 28th of May and the 2nd of June, 1914, attempts to fly were made. After acquiring speed by running on hydroplane floats on the surface of Lake Keuka the machine lifted into the air several different times. The longest time off the water with the Langley motor was approximately five seconds. Dr. Zahm stated that "it was apparent that owing to the great weight which had been given to the structure by adding the floats it was necessary to increase the propeller thrust." So no further attempts were made to fly with the Langley 52 hp. engine.

It is to be regretted that the Institution published statements repeatedly** to the effect that these experiments of 1914 demonstrated that Langley's plane of 1903 without essential modification was the first heavier-than-air machine capable of maintaining sustained human flight.

As first exhibited in the United States National Museum, January 15, 1918, the restored Langley plane of 1903 bore the following label:

THE ORIGINAL, FULL-SIZE
LANGLEY FLYING MACHINE, 1903

For this simple label others were later substituted containing the claim that Langley's machine "was the first man-carrying aeroplane in the history of the world capable of sustained free flight."

Though the matter of the label is now an issue, it seems only fair to the Institution to say that in September 1928, Secretary Abbot finally caused the label of the Langley machine to be changed to read simply as follows:

LANGLEY AERODROME

THE ORIGINAL SAMUEL PIERPONT LANGLEY FLYING MACHINE OF 1903, RESTORED.

Deposited by
The Smithsonian Institution

301,613

This change has frequently been overlooked by writers on the controversy.

In January 1942, Mr. Fred C. Kelly, of Peninsula, Ohio, communicated to me a list of differences between the Langley plane as tested in 1914 and as tested in 1903, which he had received from Dr. Wright. This list is given verbatim below. The Institution accepts Dr. Wright's statement as correct in point of facts. Inferences from the comparisons are primarily the province of interested experts and are not discussed here.

* For an account of early Langley and Wright aeronautical investigations, see *Smithsonian Report* for 1900 and *The Century Magazine* of September 1908.

† Reprinted from the *Smithsonian Miscellaneous Collections*, Vol. 103, No. 8, October 24, 1942 (Publication 3699).

‡ Secretary.

** *Smithsonian Reports:* 1914, pp. 9, 219, 221, 222; 1915, pp. 14, 121; 1917, p. 4; 1918, pp. 3, 28, 114, 166. *Report of U. S. National Museum*, 1914, pp. 46 and 47.

COMPARISON OF THE LANGLEY MACHINE OF 1903 WITH THE HAMMONDSPORT MACHINE OF MAY–JUNE, 1914

	Langley, 1903	Hammondsport, 1914
	Wings	
1	SIZE: 11′6″ × 22′6″ (L.M. p. 206).	SIZE: 10′11 3/4″ × 22′6″.
2	AREA: 1040 sq.ft. (L.M. p. 206).	AREA: 988 sq.ft.
3	ASPECT RATIO: 1.96.	ASPECT RATIO: 2.05.
4	CAMBER: 1/12 (L.M. p. 205).	CAMBER: 1/18.
5	LEADING EDGE: Wire 1/16″ diameter (L.M. pl. 66).	LEADING EDGE: Cylindrical spar 1 1/2″ dia. at inner end, tapering to 1″ dia. at outer end.
6	COVERING: Cotton fabric, not varnished.	COVERING: Cotton fabric, varnished.
7	CENTER SPAR: Cylindrical wooden spar, measuring 1 1/2″ dia. for half its length and tapering to 1″ at its tip (L.M. p. 204). Located on upper side of wing.	CENTER SPAR: Cylindrical spar about 1 1/2″ dia. at inner end, tapering to about 1″ dia. at outer end. Located on upper side of wing. This center spar was reinforced (1) by an extra wooden member on the under side of the wing, which measured 1″ × 1 1/2″ and extended to the 7th rib from the center of the machine; and (2) by another wooden reinforcement on the under side extending out about one-fourth of the length of the wing.
8	RIBS: Hollow box construction (L.M. pls. 66, 67).	RIBS: Most of the original Langley box ribs were replaced with others made at Hammondsport (Manly letter, 1914). The Hammondsport ribs were of solid construction and made of laminated wood. That part of the rib in front of the forward spar was entirely omitted.
9	LOWER GUY-POSTS: A single round wooden post for each pair of wings (see Fig. 3), 1 1/4″ in dia. 6 1/2′ long (L.M. pl. 62, p. 184).	LOWER GUY-POSTS: Four for each pair of wings (see Fig. 4), two of which were of streamline form measuring 1 1/4″ × 3 1/2″ × 54″ long; and two measuring 2″ × 2″ with rounded corners, 3′9″ long.
10	The front wing guy-post was located 28 1/2″ in front of the main center spar (L.M. pl. 53).	The front wing guy-posts were located directly underneath the main center spar, 28 1/2″ further rearward than in 1903.
11	The rear wing guy-post was located 31 1/2″ in front of the main center spar (L.M. pl. 53).	The rear wing guy-posts were located directly under the main center spar, 31 1/2″ further rearward than in 1903.
12	UPPER GUY-POSTS: For each pair of wings a single steel tube 3/4″ dia., 43″ long (L.M. p. 184, pl. 62).	UPPER GUY-POSTS: For each pair of wings, two streamline wooden posts each 1 1/4″ × 3 1/2″, 76″ long, forming an inverted V. (See Fig. 4.)
13	Front wing upper guy-post located 28 1/2″ in front of the main center spar (L.M. pl. 53).	Front wing upper guy-posts located directly over main spar, 28 1/2″ further rearward than in 1903.
14	The rear wing upper guy-post was located 31 1/2″ in front of the main center spar (L.M. pl. 53).	The rear wing guy-posts were located directly over the main center spar, 31 1/2″ further rearward than in 1903.
15	TRUSSING: The wing trussing wires were attached to the spars at the 5th, 7th and 9th ribs out from the center (L.M. pl. 54). The angles between these wires and the spars to which they were attached are shown in Fig. 3.	TRUSSING: A different system of wing trussing was used, and the wing trussing wires were attached to the spars at the 3rd, 6th and 9th ribs from the center. The angles between these wires and the spars to which they were attached were all different from those in the original Langley machine. (See Fig. 4.)

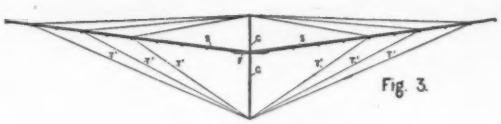

Fig. 3.

LANGLEY WING TRUSSING 1903.

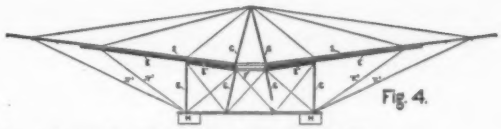

Fig. 4.

HAMMONDSPORT WING TRUSSING 1914.

Control Surfaces

16 VANE RUDDER: A split vane composed of two surfaces united at their leading edges and separated 15″ at their trailing edges, thus forming a wedge. Each surface measured 2′3″ × 4′6″, with aspect ratio .5 (L.M. p. 214, pls. 53, 54).

VERTICAL RUDDER: The Langley vane rudder was replaced by a single plane vertical rudder which measured 3′6″ × 5′, with aspect ratio of .7.

17 Operated by means of a wheel located slightly in front of the pilot at his right side and at the height of his shoulder (L.M. p. 216, pls. 53, 54).

Operated at Hammondsport through the Curtiss steering wheel in some tests (Zahm affidavit pp. 5, 6), through the Curtiss shoulder yoke in some others (Manly letter, 1914), and fixed so as not to be operable at all in still others (Zahm affidavit p. 7).

18 Used for steering only (L.M. p. 214).

Used "as a vertical aileron to control the lateral poise of the machine" (Zahm affidavit p. 6), as well as for steering (Zahm affidavit p. 7).

19 PENAUD TAIL: This was a dart-shaped tail having a vertical and a horizontal surface (Penaud tail), each measuring 95 sq.ft. It was located in the rear of the main frame.

TAIL RUDDER: Same size and construction as in 1903.

20 Attached to a bracket extending below the main frame.

Attached to same bracket at a point about 8″ higher than in 1903.

21 "Normally inactive" (L.M. p. 216) but adjustable about a transverse horizontal axis by means of a self-locking wheel located at the right side of the pilot, even with his back, and at the height of his shoulder (L.M. pls. 51, 53).

Operable about a transverse horizontal axis and connected to a regular Curtiss elevator control post directly in front of the pilot (Zahm affidavit p. 5).

22 Immovable about a vertical axis (L.M. p. 214, pl. 56, Fig. 1). No means were provided for adjusting this rudder about a vertical axis in flight. "Although it was necessary that the large aerodrome should be capable of being steered in a horizontal direction, it was felt to be unwise to give the Penaud tail and rubber motion in the horizontal plane in order to attain this end" (L.M. p. 214).

Immovable about a vertical axis on May 28, 1914, only. Thereafter it was made movable about a vertical axis and was connected through cables to a Curtiss steering wheel mounted on a Curtiss control post directly in front of the pilot.

23 KEEL: A fixed vertical surface underneath the main frame measuring 3′2″ in height by 6′ average length. Area 19 sq.ft. (L.M. pl. 53).

KEEL: Entirely omitted.

System of Control

24 LATERAL STABILITY: The dihedral only was used for maintaining lateral balance (L.M. p. 45).

LATERAL STABILITY: Three means were used for securing lateral balance at Hammondsport: The dihedral angle as used by Langley, a rudder which "serves as a vertical aileron" (Zahm affidavit p. 6), and the Penaud tail rudder. The last two constituted a system "identical in principle with that of Complainant's [Wright] combined warping of the wings and the use of the vertical rudder." (Zahm affidavit p. 6.)

25	LONGITUDINAL STABILITY: Langley relied upon the Penaud system of inherent stability for maintaining the longitudinal equilibrium. "For the preservation of the equilibrium [longitudinal] of the aerodrome, though the aviator might assist by such slight movements as he was able to make in the limited space of the aviator's car, the main reliance was upon the Penaud tail" (L.M. p. 215).	LONGITUDINAL STABILITY: At Hammondsport the Penaud inherent longitudinal stability was supplemented with an elevator system of control.
26	STEERING: Steering in the horizontal plane was done entirely by the split-vane steering rudder located underneath the main frame (L. M. p. 214).	STEERING: On one day, May 28, 1914, steering in the horizontal plane was done with the vertical rudder which had been substituted for the original Langley split-vane steering rudder. After May 28th the steering was done by the vertical surface of the tail rudder (Zahm affidavit p. 7), which in 1903 was immovable about a vertical axis (L.M. p. 214).
	Power Plant	
27	MOTOR: Langley 5 cylinder radial.	MOTOR: Langley motor modified.
28	IGNITION: Jump spark with dry cell batteries (L.M. p. 262).	IGNITION: Jump spark with magneto.
29	CARBURETOR: Balzer carburetor consisting of a chamber filled with lumps of porous cellular wood saturated with gasoline. The air was drawn through this wood. There was no float feed (L.M. p. 225).	CARBURETOR: Automobile type with float feed.
30	RADIATOR: Tubes with radiating fins.	RADIATOR: Automobile radiator of honeycomb type.
31	PROPELLERS: Langley propellers (L.M. pl. 53, pp. 178–182).	PROPELLERS: Langley propellers modified "after fashion of early Wright blades."
	Launching and Floats	
32	LAUNCHING: Catapult mounted on a houseboat.	LAUNCHING: Hydroplanes, developed 1909–1914, attached to the machine.
33	FLOATS: Five cylindrical tin floats, with conical ends, attached to underside of main frame at appropriate points, and about six feet above lowest part of machine.	FLOATS: Two wooden hydroplane floats, mounted beneath and about 6 feet to either side of the center of the machine at the lateral extremities of the Pratt system of trussing used for bracing the wing spars of the forward wings; and one (part of the time two) tin cylindrical floats with conical ends, similar to but larger than the Langley floats, mounted at the center of the Pratt system of trussing used for bracing the rear wings. All of the floats were mounted from four to five feet lower than the floats of the original Langley, thus keeping the entire machine above the water.
	Weight	
34	TOTAL WEIGHT: With pilot 850 pounds (L.M. p. 256).	TOTAL WEIGHT: With pilot, 1170 pounds.
35	CENTER GRAVITY: $^3/_8$" above line of thrust.	CENTER GRAVITY: About one foot below line of thrust.

Since I became Secretary, in 1928, I have made many efforts to compose the Smithsonian-Wright controversy, which I inherited. I will now, speaking for the Smithsonian Institution, make the following statement in an attempt to correct as far as now possible acts and assertions of former Smithsonian officials that may

have been misleading or are held to be detrimental to the Wrights.

1. I sincerely regret that the Institution employed to make the tests of 1914 an agent who had been an unsuccessful defendant in patent litigation brought against him by the Wrights.

2. I sincerely regret that statements were repeatedly made by officers of the Institution that the Langley machine was flown in 1914 "with certain changes of the machine necessary to use pontoons," without mentioning the other changes included in Dr. Wright's list.

3. I point out that Assistant Secretary Rathbun was misinformed when he stated that the Langley machine "without modification" made "successful flights."

4. I sincerely regret the public statement by officers of the Institution that "The tests" [of 1914] showed "that the late Secretary Langley had succeeded in building the first aeroplane capable of sustained free flight with a man."

5. Leaving to experts to formulate the conclusions arising from the 1914 tests as a whole, in view of all the facts, I repeat in substance, but with amendments, what I have already published in Smithsonian Scientific Series, Vol. 12, 1932, page 227:

> The flights of the Langley aerodrome at Hammondsport in 1914, having been made long after flying had become a common art, and with changes of the machine indicated by Dr. Wright's comparison as given above, did not warrant the statements published by the Smithsonian Institution that these tests proved that the large Langley machine of 1903 was capable of sustained flight carrying a man.

6. If the publication of this paper should clear the way for Dr. Wright to bring back to America the Kitty Hawk machine to which all the world awards first place, it will be a source of profound and enduring gratification to his countrymen everywhere. Should he decide to deposit the plane in the United States National Museum, it would be given the highest place of honor, which is its due.

Durand's "Aerodynamic Theory"

There is in process a project for the reprinting of Durand's "Aerodynamic Theory" in a form which duplicates as nearly as possible the original size and binding. A committee which is composed of Dr. Theodore von Kármán, Dr. Clark B. Millikan and Mr. E. W. Robischon of the Aeronautics Department of the California Institute of Technology has been organized to direct this nonprofit venture. Subscriptions have been requested for these reprints on the basis of $50 per set with a $10 deposit accompanying the order. It was found that the sale of 100 sets would pay all expenses necessary for printing, binding and distribution. The demand for these volumes has, however, now made it possible to decrease this cost through increased distribution whereby if 200 subscribers are obtained the cost can be reduced to $30 per set. To date, seventy-five subscriptions have been obtained on the $50 plan and promises of purchase at $30 have increased the number to 150. Persons interested in purchasing this set on the basis of $10 with order and $20 upon delivery should contact the "Durand Reprinting Committee," c/o Aeronautics Department, California Institute of Technology, Pasadena, California. This should be done as soon as possible as the committee wishes to have the reprinting completed and available for distribution by the first of February.

Graphic Determination of Distance in Accelerated Airplane Motion

G. A. MOKRZYCKI

Institute of Technology, Warsaw

CONSIDER AN AIRPLANE moving along the x axis, then if

$$(W/g)(dv_x/dt) = \Gamma \tag{1}$$

the work done by the force Γ is equal to the change in kinetic energy of the airplane. Hence, for air density ρ = constant

$$\Gamma ds = dE = d(WV^2/2g) = (W/g)V dV \tag{2}$$

where W is the weight of the airplane and s is the distance. With dynamic pressure $q = \rho(V^2/2)$. Eq. (2) becomes

$$\Gamma ds = (W/\rho g)dq \tag{3}$$

$$s = (W/\rho g)\int_{q_1}^{q_2}(1/\Gamma)dq \tag{4}$$

Eq. (4) may be represented graphically as shown in Fig. 1. By plotting $\Gamma = f(q)$ and then $1/\Gamma = \varphi(q)$, the area A can be determined between the limits q_1 and q_2 giving the solution

$$s = (W/\rho g)A \tag{5}$$

This principle may be easily adapted to the computation of take-off, landing, diving, etc.

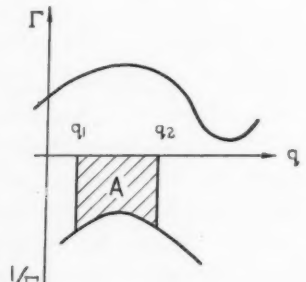

FIG. 1. Analysis of accelerated motion.

EXAMPLES

Take-Off Distance

(*a*) *Landplane.* The equation of motion is

$$\Gamma = T - D - f(W - L) = T - fW - (C_D - fC_L)Sq \tag{6}$$

where T = thrust, D = drag, f = coefficient of friction, and L = lift. For take-off, the angle of attack, hence C_D and C_L, are known. The final value of q necessary to carry the weight of the airplane is

$$q_t = W/C_{L_{max.}}S$$

The relation $T = f(q)$ is also given for a given propeller. It can often be considered with good approximation as a straight line. In accordance with Eq. (6), $-fW$ is plotted against q (see Fig. 2) and from this line $-(C_D - fC_L)Sq$ is also plotted. Thus Γ is determined as a function of q as the intercept between the curves of T and $-(C_D - fC_L)Sq$. Plotting $1/\Gamma = \varphi(q)$ and integrating graphically between the limits $q_1 = 0$ and $q_2 = q_t$ gives the solution

$$s = (W/\rho_0 g)A_t$$

where ρ_0 is the density at sea level.

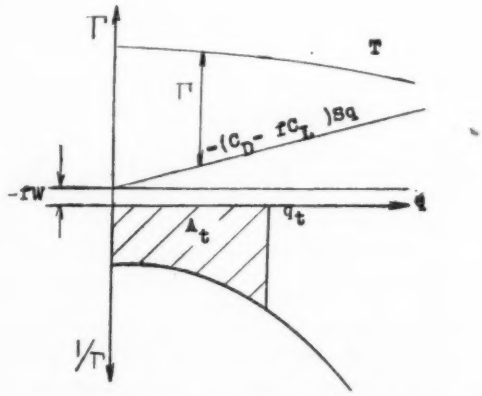

FIG. 2. Determination of take-off distance for a landplane.

(*b*) *Seaplane.* The equation of motion is

$$\Gamma = T - D - D_w \tag{7}$$

where D_w is the water drag, given as a function of q by towing tank tests. Similar plotting as shown in Fig. 3 gives the desired solution.

Landing Run

(*a*) *Landplane.* If $T = 0$, the equation of motion is

$$\Gamma = -D - f(W - L) = -fW - (C_D - fC_{L_{max.}})Sq \tag{8}$$

in which $C_{L_{max.}}$, the corresponding C_D, and also the landing speed or dynamic pressure, $q_l = W/SC_{L_{max.}}$,

Received April 2, 1942.

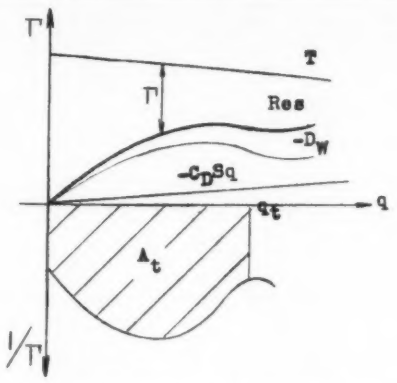

FIG. 3. Determination of take-off distance for a seaplane.

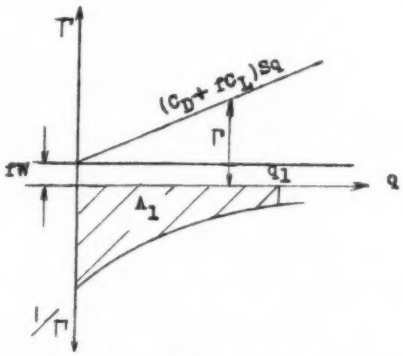

FIG. 4. Determination of landing run for a landplane.

are known. The plotting according to Eq. (8) is shown in Fig. 4, the limits of integration being $q_1 = q_t$ and $q_2 = 0$. The seaplane is handled in a similar manner.

Diving or Zoom

(*a*) *Constant Density.* If the density is considered as constant and it is desired to determine the distance necessary for the airplane to attain some given velocity V_2, the equation of motion may be written as

$$\Gamma = \pm T - C_D S q \mp W \sin \gamma \qquad (9)$$

where γ is the inclination of the flight path. The fact that the propeller may produce either propulsive or braking action is taken care of by the double sign on the thrust term. The solution is carried out graphically as before, it being possible to take into account variations of C_D with q or basically with Reynolds Number.

(*b*) *Variable Density.* When the change in atmospheric density is considered, $\rho_z = f(s)$, the integration must be carried out by a step-by-step process as shown in Fig. 5. Eq. (9) is again the equation of motion, the relation $C_D = f(\text{R.N.}) = F(q)$ being given by wind-tunnel test. Now the distance s must be divided into a sufficient number of parts Δs, so that ρ_z may be considered as constant for each part. If the density is constant and equal to the sea level value ρ_0, the distance is

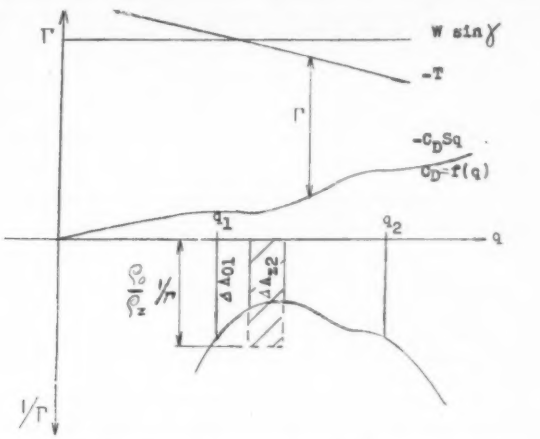

FIG. 5. Determination of diving distance with varying density.

$$s = \frac{W}{\rho_0 g}\int_{q_1}^{q_2} \frac{1}{\Gamma}\, dq = \frac{W}{\rho_0 g}\sum(\Delta A_{0i})$$

while with a variable density ρ_z, the distance is

$$s = \frac{W}{\rho_0 g}\int_{q_1}^{q_2} \left(\frac{\rho_0}{\rho_z}\right)\frac{1}{\Gamma} dq \qquad (10)$$

After plotting $1/\Gamma = \varphi(q)$, the area A is divided into several parts ΔA_{0i}. The small areas or increments of distance $\Delta s_{0i} = (W/\rho_0 g)(\Delta A_{0i})$ are then increased in the ratio ρ_0/ρ_z in accordance with Eq. (10). This calculation determines the value of Δs_1 and allows the value of the density ρ_{z_2} for the next increment of distance, Δs_2, to be computed. The process is repeated until the given value of q is reached or the value of q corresponding to $\Gamma = 0$—that is, the limiting or terminal velocity—is attained.

SECOND APPROXIMATION

If necessary, a second approximation can be obtained. Knowing the values of q, or velocities corresponding to the heights z, or density ρ_z, the corresponding values of Reynolds Number can be determined, from which a new curve C_DSq against q can be plotted, where $C_D = F/(\text{R.N.})$

This new curve will determine new values of the resultant force Γ_R. By plotting $1/\Gamma_R$ against q and repeating the procedure, more exact values of the distance s can be obtained.

Pendulum-Type Vibration Absorbers

R. G. MANLEY

Middlesex, England

Letter to the Editor on "Optimum Performance of Pendulum-Type Torsional Vibration Absorbers," published in the JOURNAL OF THE AERONAUTICAL SCIENCES, July, 1942.

SUMMARY

The volume of literature dealing with the theory of pendulum-type absorbers has grown steadily since the study of these devices was first put on a rational basis by Stieglitz.[1] A notable description of the characteristics of absorbers has been given by Zdanowitch and Wilson,[2] and the application to simple two- and three-mass systems (as well as to more complicated engine combinations) has been fully treated by Ker Wilson.[3] Most authorities are now agreed that the simplest and most direct method of analysis is by use of the concept of "effective inertia," an introduction to which is given in reference 4. Any other method of attack is, of course, acceptable so long as the results are valid, but in the case of the paper under consideration an initial assumption nullifies the argument.

The solution by normal mathematical analysis is given below, followed by a confirmatory solution by the effective-inertia method. By these means an expression is obtained for the (single) resonant frequency at any given order (ratio of forced frequency to rotational speed), and it is shown that the corresponding expression derived from Mr. Shieh's paper is inaccurate. The present contribution concludes with an inquiry into the source of error.

ERRORS INVOLVED

For quantitative purposes an engine-propeller combination cannot be regarded as a simple two-mass system, since it is now quite clear that the effect of propeller flexibility cannot be ignored. But even if this consideration is set aside, there is a formal error in Mr. Shieh's analysis. The torsional pendulum to which he reduces the two-mass system is not equivalent in all respects to the original system, since the important rolling component of angular motion is eliminated by the reduction.

The only considerations that can usefully be employed in the determination of optimum pendulum designs are the restriction of pendulum amplitudes to small values and the provision of adequate manufacturing tolerances. On these counts the indication is that the pendulum should be designed so as to include as much as possible of the material of the balance-weight of which it forms a part.

SOLUTION BY NORMAL METHODS

Retaining the original two-mass system, let a torque $M = M_0 \cos(pt - \psi)$ act on I_2. Using the same notation as in the paper, but denoting the angular displacements of I_1 and I_2 by θ_1 and θ_2, the following equations of motion are obtained:

$$\begin{aligned} [J_0 + m(R+L)^2 + mk^2]\ddot{\theta}_2 + mL(R+L)\ddot{\varphi} &= M - K(\theta_2 - \theta_1) \\ (R+L)\ddot{\theta}_2 + L\ddot{\varphi} &= -R\Omega^2\phi \\ I_1\ddot{\theta}_1 &= K(\theta_2 - \theta_1) \end{aligned} \tag{1}$$

where $J_0 = I_2 - m(R+L)^2$. The first and second of these equations correspond to Eqs. (9) and (10) of the paper, and the third is simply the equation of motion for I_1. Substituting $\theta_2 = \theta_m \cos(pt - \psi)$, with similar expressions for θ_1 and ϕ, and solving Eq. (1) for θ_m, for $mk^2 \ll I_2$:

$$\frac{(K - I_1p^2)(K - I_2p^2) - K^2}{K - I_1p^2} - \frac{mp^4L(R+L)^2}{R\Omega^2 - Lp^2} = \frac{M_0}{\theta_m} \tag{2}$$

Now let

$$n^2 = R/L,\ \Omega_n^2 = \frac{K(I_1 + I_2)}{I_1I_2},\ \mu = m(R+L)^2\Omega_n^2/K,\ I_2 = UI_1,$$

$$g = \frac{p}{\Omega_n} = \frac{\text{frequency of forced vibration}}{\text{natural frequency for } \mu = 0}$$

and $N = P/\Omega$ = order of forced vibration; then Eq. (2) becomes

$$\frac{g^2 - 1}{[U/(1+U)] - g^2} - \frac{\mu N^2}{(1+U)(n^2 - N^2)} = \frac{M_0}{p^2UI_1\theta_m} \tag{3}$$

If $N = n$, θ_m is zero, so that the nth order input energy is absorbed; but for any other value of the ratio N/n, the value g_r of g for which θ_m is infinite is given by Eq. (3) as:

$$g_r = \sqrt{\frac{(1+U) - (N/n)^2[1 + U - [U\mu/(1+U)]]}{1 + U - (N/n)^2(1 + U - \mu)}} \tag{4}$$

For any particular order N there is therefore only one resonant frequency: $g_r\Omega_n/2\pi$ c.p.s.

SOLUTION BY EFFECTIVE-INERTIA METHOD

The inertia of the pendulum carrier is $I_2 - m(R+L)^2 = UI_1[1 - [\mu/(1+U)]]$, and the effective inertia of the pendulum is[2, 3]

$$\frac{m(R+L)^2}{1 - N^2/n^2} = \frac{U\mu I_1}{(1+U)[1 - (N^2/n^2)]}$$

The total effective inertia at the carrier is therefore

$$J_e = UI_1\left\{1 + \frac{\mu N^2}{(1+U)n^2[1 - (N^2/n^2)]}\right\} \tag{5}$$

The resonant frequency $p_r/2\omega$ is given by

$$p_r^2 = K(I_1 + J_e)/I_1J_e$$

and, putting $p_r = g_r\Omega_n$, Eq. (4) is confirmed.

ANALOGOUS EXPRESSION DERIVED FROM THE PAPER

The equation

$$(1 - \mu)g_r^4 - (1 + f^2n^2)g_r^2 + f^2n^2 = 0$$

given in the paper and purporting to determine the resonant frequencies can be reduced to the same form as Eq. (4) above, for

$$f = \Omega/\Omega_n,\ g_r = p_r/\Omega_n, \text{ and } N = p_r/\Omega$$

Hence, $f = g_1/N$, and the formula is obtained:

$$g_r = \sqrt{\frac{1 - (N^2/n^2)}{1 - [(1-\mu)N^2/n^2]}} \tag{6}$$

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Comparison of Results

On comparing Eqs. (4) and (6), in both of which μ has the same meaning, it is evident that there is a difference in essential form. For given values of shaft stiffness K and "reduced" inertia $I = I_1I_2/(I_1 + I_2)$, the resonant frequency ratio as given by Eq. (6) is dependent merely on μ and the ratio N/n, whereas Eq. (4) involves the inertia ratio $U = I_2/I_1$ as well.

In Fig. 1 the full and dotted lines show the curves of resonant frequency ratio g_r plotted against order number $N = p/\Omega$ for the particular case in which $n = 4.5$ (i.e., the pendulum is tuned to the 4.5th order, so that $R/L = 4.5^2$), $U = 0.2$ and $\mu = 0.339$; the full line conforms to Eq. (4) and the dotted line to Eq. (6). Both curves show that there is a range of values of N for which there is no resonant frequency (g_r^2 negative). Eq. (4) of this analysis gives the range (AA′) as $4.61 < N < 5.31$, whereas Eq. (6) derived from Mr. Shieh's paper gives the range (BB′) as 4.5 lb N lb 5.535. Thus, whereas the resonant frequency ratio for the 5.5th order is 2.43, as denoted by the point P, according to Eq. (6) there is no resonance, as the value $N = 5.5$ lies within the range BB′ for which g_r^2 is negative.

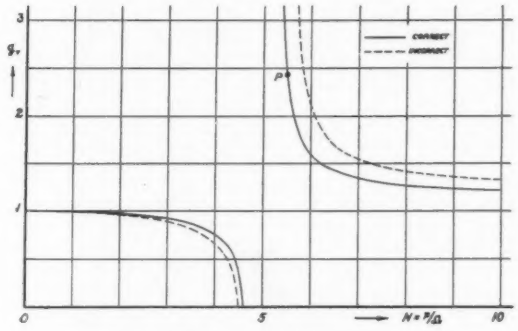

Fig. 1. Variation g_r with order number "N."

Optimum Performance

Turning now to a consideration of the best value of μ for any particular case, it must be emphasized at once that any attempt to avoid resonant conditions will prove unsuccessful if the effect of the propeller flexibility is neglected. Assuming, however, that the propeller is a rigid inertia, Fig. 2 shows the variation with μ of the resonant frequency of the 9th order for the particular case taken in section V, and with a natural frequency of 3,300 c.p.m. when $\mu = 0$. This diagram shows clearly how inaccurate any results based upon Eq. (6) can be. The highest value of μ included is 0.6, since as a practical fact it is not possible to make the inertia $m(R + L)^2$ of the pendulum greater than about half the total inertia at the crank and, with $U = 0.2$, the corresponding limit for μ is approximately 0.6. Up to this limit it is certainly true, as Mr Shieh implies, that the 9th order resonance is well below operating speed range, but his statement that "the system is constantly in resonance with an unimportant order of torque harmonic at all engine speeds for any particular value of μ" is invalidated by the fact that the propeller cannot be regarded as a rigid inertia. Furthermore, since propeller-excited torsional orders are often encountered and in further development may even be troublesome, there will be resonances with a number of orders which cannot be considered "unimportant."

Since the resonance characteristics of engine-propeller systems are difficult to determine theoretically, the only considerations that can usefully be employed to determine pendulum sizes are the restriction of pendulum amplitudes to small values and the provision of adequate manufacturing tolerances. Both these considerations indicate that μ should be made as large as possible.[3] In practical cases μ has no real significance, since the propeller has no unique inertia value I_1, but the fact remains that the most efficient absorber is that in which as much as possible of the balance-weight is included in the pendulum.

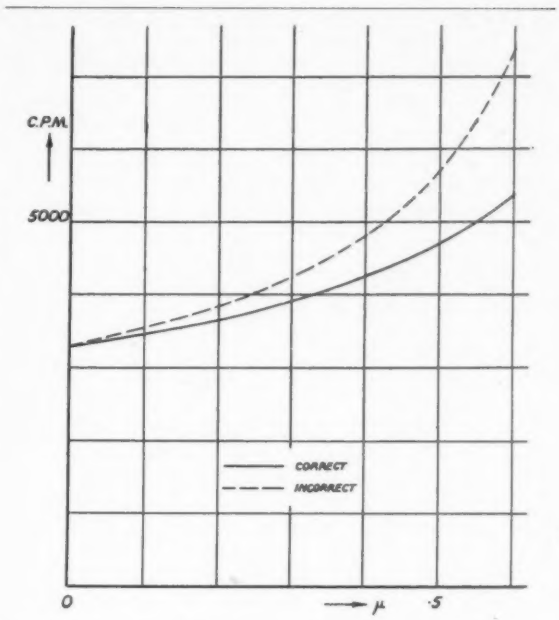

Fig. 2. Variation of 9th order natural frequency with μ = absorber parameter.

Source of Error

The reason for the erroneous nature of Mr. Shieh's results lies in the initial assumption that his "reduced" system (Fig. 2 in the paper) corresponds in all particulars to the original two-mass system. In actual fact the only factor common to the two systems is the natural frequency. Reduction of a two-mass system to a torsional pendulum has the effect of eliminating the rolling motion, and the initial error is contained in the statement that "only relative motion is involved." With a single-row radial engine combination, every order of forced vibration, whether excited by engine forces or by propeller disturbances, has a rolling component of angular motion, the effect of which is to render the position of the actual node in the shaft dependent on the forced frequency. It would be possible to reduce a true two-mass system (i.e., one with a rigid inertia in place of the propeller) satisfactorily to a torsional pendulum in which the inertia is I_2 and the shaft stiffness is that of the appropriate nodal length, but there is no advantage to be gained by such a procedure. The formal error in the analysis set forth in the paper is caused by the fact that the resemblance between the systems of Figs. 1 and 2 begins and ends with the possession of the same natural frequency.

References

[1] Stieglitz, A., *Influence of Pendulum Masses on Torsional Vibration*, D.V.L. Yearbook, Vol. II, 1938.

[2] Zdanowitch, R. W., and Wilson, T. S., *Elements of Pendulum Dampers*, Proc. I. Mech. E., June, 1940.

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[4] Manley, R. G., *Fundamentals of Vibration Study*, Chapter IV; Chapman & Hall, London, 1942.

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